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Development of Probabilistic Design Methodology for Composite Structures

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16. Abstract

This program was conducted by Vought Aircraft Company, Dallas, Texas, to perform technical studies to aid in the development of a probabilistic design methodology. The foundation of the probabilistic design approach, applied to composite structure, is to base design criteria and objectives on reliability targets instead of factors of safety. Control of the process, in terms of how much it differs from the traditional approach, is maintained by the "Probability of Structural Failure." The key technical issues addressed in this contract were the overall assessment of the accuracy of the methodology, current reliability experience, definition of appropriate goals, and database development. The overall assessment of the accuracy of the methodology was done by reviewing current published documents and papers in the probabilistic design field. This review focused on similarities and differences between approaches. The database development was done by visiting airline maintenance depots and naval aviation depots to collect data on structural failures. The analyses of such data produced historical values for aircraft structural reliability. Current structural reliability issues and reliability goals were addressed by analyzing the wing box of the Lear Fan aircraft using Vought's Probabilistic Design Model. Measures of structural reliability such as single flight hour probability of failure for the whole wing box, including upper skin, lower skin, and substructure were produced.

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EXECUTIVE SUMMARY

This program was conducted by Northrop Grumman Commercial Aircraft Division (NGCAD), Dallas, Texas, to provide an evaluation of the application of the probabilistic design methodology to composite aircraft structures. The probabilistic design approach is based on design criteria involving the use of reliability targets rather than deterministic factors of safety. Control of the process, in terms of how much it differs from the traditional approach, is maintained by quantification of the reliability target as expressed by the probability of structural failure. The benefits of probabilistic design are twofold: quantification of the structural reliability and the ability to manage the risk through the identification of important design drivers. The key technical issues addressed in this contract were the overall assessment of the accuracy of the methodology, current reliability experience, and definition of appropriate goals and database development.

The overall assessment of the accuracy of the methodology was done by reviewing current published documents and papers in the probabilistic design field. This review focused on similarities as well as differences between approaches.

The database was developed by visiting American Airlines, Delta Airlines, United Airlines and NADEP, North Island, to collect data on structural failures. Information was also obtained from De Havilland, Toronto. The analyses of such data provided historical values for aircraft structural reliability.

A structural reliability analysis case study was conducted on the wing box of the Lear Fan aircraft using Northrop Grumman's Probabilistic Design Model. Measures of structural reliability (single-flight probability of failure) for the upper wing skin, lower wing skin, and wing substructure as well as wing box as a whole were produced. These reliabilities were also grouped by failure mode. This analysis was performed to give insight on how the Probabilistic Design Model would predict the reliability values (as designed) in modern day aircraft. For the restricted (3.29 g) Lear Fan aircraft, the single-flight probability of failure was predicted to be 1.2×10^{-9}.

1. INTRODUCTION.

A knowledge of the inherent risk of failure in the design of any component or product is becoming increasingly important to both the manufacturer and customer. Designers and management must concern themselves with the ability to assess risk, identify parameters which drive the risk, and minimize the risk given other program constraints. In the case of aircraft design, current practice involves minimizing risk of structural failure by the application of safety factors and judicious use of material properties.

It is recognized that the number of aircraft accidents attributable to structural airframe component failure has been very low in the past few decades. Yet designs yielding an unknown risk pose a number of problems for the future. As designs grow more critical and competitive and with increasing emphasis on warranties, there will be a growing need to assess and optimize reliability. New aircraft are departing dramatically from traditional environments (e.g., reusable launch vehicle, high-speed civil transport), where application of safety factors may not be sufficient to provide adequate safety. In the case of the design of composite airframe components in particular, relatively large knockdown factors are employed to account for uncertainties, resulting in a substantial weight increase without a quantifiable measure of the effect on structural reliability.

Analysis of aircraft structure using probabilistic methods provides a tool for meeting these needs. All design parameters are treated as variables, and the basic result from the analysis is a probability of failure or risk. Specifically, the probabilistic structural analysis methodology is capable of yielding output such as (1) safety (risk) quantification, (2) design variable sensitivity analysis, (3) cost and weight reduction scenarios, and (4) establishment of optimum inspection intervals. Given a target probability of failure, there is a potential for designs to be optimized.

The objective of this report is to summarize current efforts pertaining to probabilistic analysis of composite structures, describe the Northrop Grumman Commercial Aircraft Division (NGCAD) composite probabilistic analysis methodology, show an example application using the NGCAD approach, and list the type and frequency of typical operations damage to composite components.

The following sections of this report contain a review of three unique industry composite structural probabilistic design methods, an overview and example application (Lear Fan 2100 wing) of the NGCAD methodology, and a summary of maintenance information (operational damage) obtained from visits to commercial airline and military composite repair facilities.

2. INDEPENDENT METHODOLOGY ASSESSMENT.

This task involved a review of current published documents in the field of probabilistic design. This review addressed the work being done by Chamis [1], Kan [2], and Rouchon [3]. This review focused on similarities and differences between the various methodologies and noted areas where they are complementary. A description of the Northrop Grumman Commercial Aircraft Division (NGCAD) probabilistic approach to composites is provided in section 3 for comparison. Figure 2-1 shows the overlapping areas of application of these methodologies.

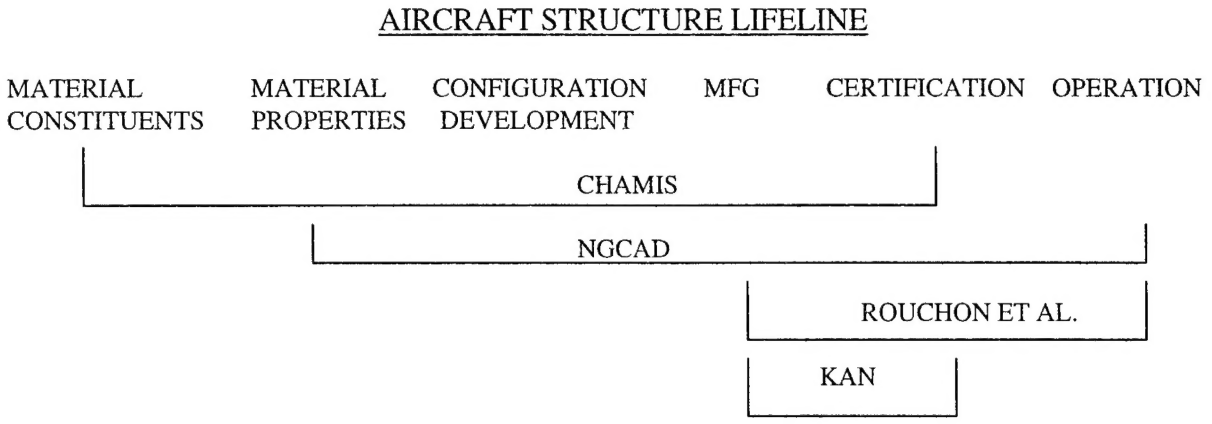


FIGURE 2-1. INDEPENDENT METHODOLOGY ASSESSMENT SUMMARY CHART

2.1 GENERAL REVIEW.

A literature search of probabilistic design methodology was done. Published documents reviewed included the recent work of Chamis, Kan, and Rouchon. Some methodologies focused on the design process and included a finite element model as an integral part. These methodologies do not require a baseline deterministic design. Other methodologies started with a baseline design and calculated the probabilistic design solution by using perturbation principals or constrained optimization techniques. Another discriminating factor among methodologies was the portion of the "lifeline" (figure 2-1) of the aircraft structure being considered. The aircraft structure lifeline starts with material constituents, followed by material properties, continues with configuration development and manufacturing, and ends with the usage life. The work of Chamis includes unique approaches to assess material characteristics by using constituent properties (fiber, resin). The work reported by Rouchon has been applied to the Airbus fleet and concentrates on establishing inspection intervals. Kan's work focuses on improving certification methods and requirements.

The various methods have been classified according to their mathematical or probabilistic approach as follows:

a. Response Surface Evaluations: Multiple Regression, ANOVA, Analysis of Residuals.

b. Limit State Equations: g-Function Derivation, Most Probable Point, Correlated Design Variables, Normal Approximations, g-Function Linearization, First and Second Order Methods, Nonlinear Fitting of g-Function.

c. Monte Carlo Simulation: Crude Monte Carlo Simulation, Efficient Monte Carlo Simulations, Adaptive Importance Sampling, Harbitz, Latin Hypercube Sampling, Stochastic Monte Carlo Simulation.

Methodologies can also be classified by their application focus. These include optimum fiber orientation, probabilistic material strength derivation, material qualification and acceptance, structure sizing, failure mode prediction, load limiting application, strength reduction versus defect size, structural certification, and effective maintainability concept identification.

2.2 REVIEW OF PREVIOUS WORK.

The literature review indicated major strength points for all the methodologies investigated. NGCAD's methodology (section 3) has the ability to handle extremely complicated situations and produce converging answers in a reasonable time. Chamis' methodology avoids the problem of measurement errors during coupon testing and does not require a baseline design to initiate analysis. Kan's methodology tackles the problem of composite residual strength and the problem of composite certification. Rouchon's methodology involves a statistical determination based on the estimation of failure risk and of service inspection intervals, given manufacturing defect and accidental damage levels.

Each of the methodologies has different constraints. NGCAD's methodology depends on a preliminary design. Chamis' methodology does not cover the operational aspects of the aircraft and depends on distributional assumptions at the fiber and resin level. Rouchon's methodology makes assumptions about reliability measures. Kan's methodology is not intended as a design tool.

2.2.1 NASA Lewis Efforts [1].

Chamis of NASA Lewis, has been instrumental in the development of probabilistic design and analysis of structure for rocket motors and turbine engine components. His early work was initiated in response to a need to quantify component reliability on the space shuttle program. Chamis has led the NASA Lewis effort to develop a probabilistic design methodology for composites. The result of this work is the Integrated Probabilistic Analysis of Composite Structures (IPACS) methodology which combines physics, mechanics, specific structure, system concepts, and manufacturing. The methodology starts with the fiber mechanical and physical properties, resin properties, and fiber placement techniques. The methodology then applies a micromechanics approach to produce a laminate theory. This is followed by a probabilistic finite

element analysis using the structural analysis. At the present time, IPACS does not include operational lifetime considerations such as maintenance induced damage or foreign object damage (FOD). IPACS does not require extensive specimen testing which is cited by many as adding to the cost of composite applications. Figure 2-2 is a schematic of the IPACS methodology.

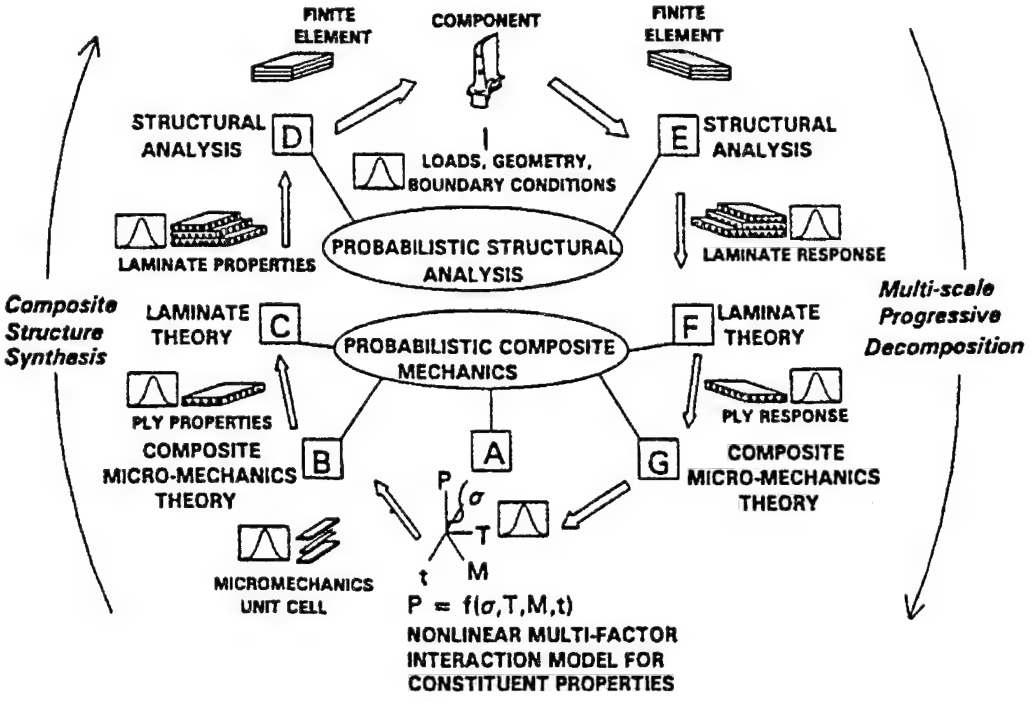


FIGURE 2-2. PROBABILISTIC DESIGN/NASA LEWIS METHODOLOGY [1]
(Schematic of the Computer Code IPACS)

Figure 2-3 provides an overall summary of the IPACS methodology as far as input, output, and application is concerned. Arrows going into the box designate an input to the model and arrows going out of the box designate either an output or a methodology application.

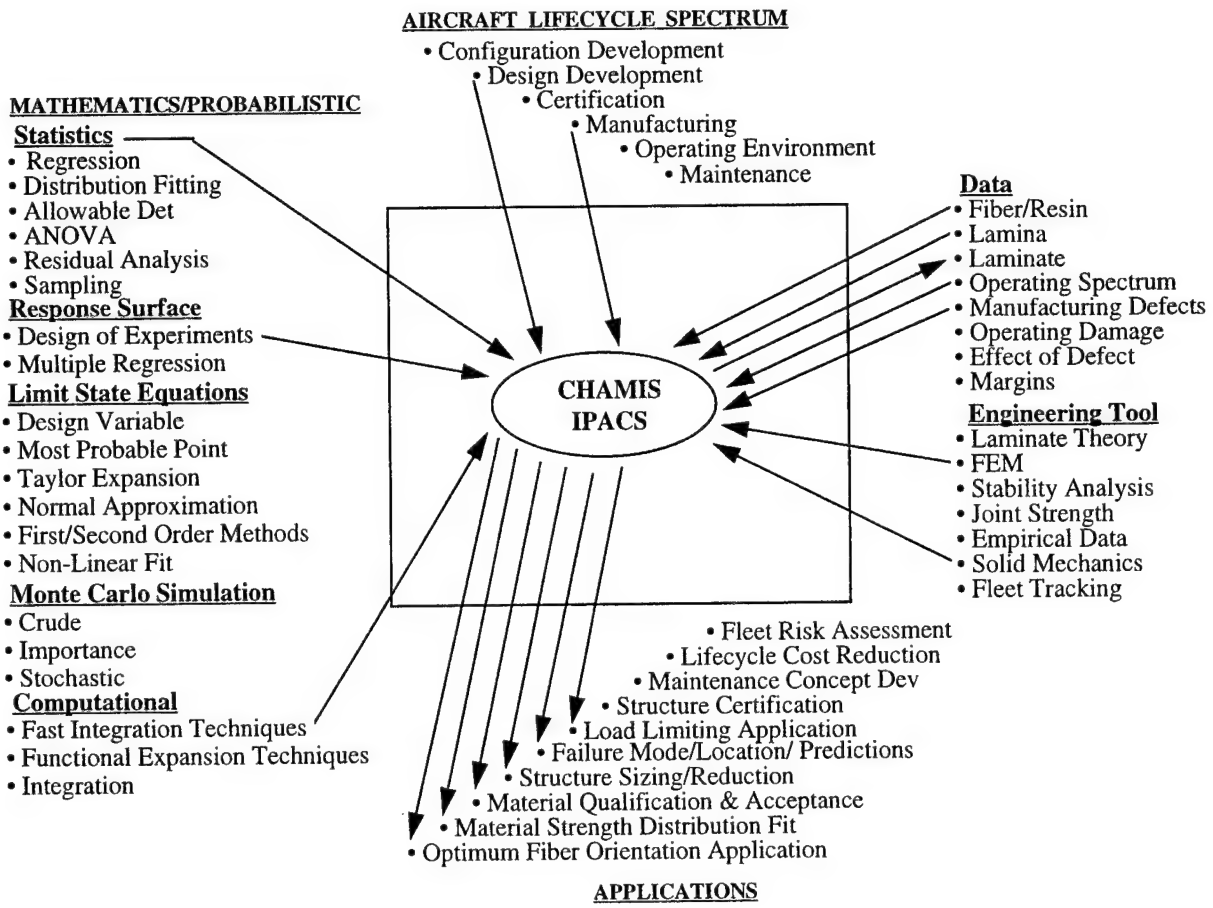


FIGURE 2-3. INDEPENDENT METHODOLOGY ASSESSMENT—CHAMIS IPACS METHODOLOGY [1]

2.2.2 Northrop Methodology [2].

The work of Kan et al., as described in [2] represents pioneering work on the requirements for static testing as well as evaluation of composite residual strength under repeated loading for composite airframe certification. Here, the Weibull analysis is applied to a large empirical database generated on DoD aircraft to evaluate structural testing requirements for achieving B-basis strength in static full-scale tests and B-basis life in full-scale fatigue tests. The statistical analysis of experimental data uses the data scatter information obtained from the DoD database to establish levels of confidence in static and fatigue test results.

In connection with certification, for required life under repeated loading, Kan et al. evaluated a number of alternative approaches including the standard Navy scatter factor approach, the use of load enhancement factor for reducing required duration of testing, the replacement of fatigue testing by increasing the ultimate strength requirement, and the introduction of variability into the spectrum. Advantages and disadvantages of each approach for verifying minimum life are established. The recommendations are that the two-parameter Weibull distribution be used to describe static strength and fatigue life data. To reduce data size requirements, data pooling by either the joint Weibull or the Sendeckyj [14] analysis is recommended. The joint Weibull

pooling technique allows for the pooling of any groups of data as long as all groups have the same shape parameter. This is done by applying the maximum likelihood estimating procedure to the groups being pooled. The mean, B-basis, and A-basis are determined for each of the individual groups as well as for the pooled data. The second pooling technique, the Sendeckyj Equivalent Strength Model [14] evaluated by Kan et al. Uses two fitting parameters to relate pooled static strength, fatigue life, and residual strength data. All three types of data are converted to equivalent static strength through the use of a wear-out equation and a fatigue power law. The equivalent static strength is then fitted to a two-parameter Weibull distribution. Figure 2-4 lists the input, output, and areas of application of the Northrop [2] methodologies.

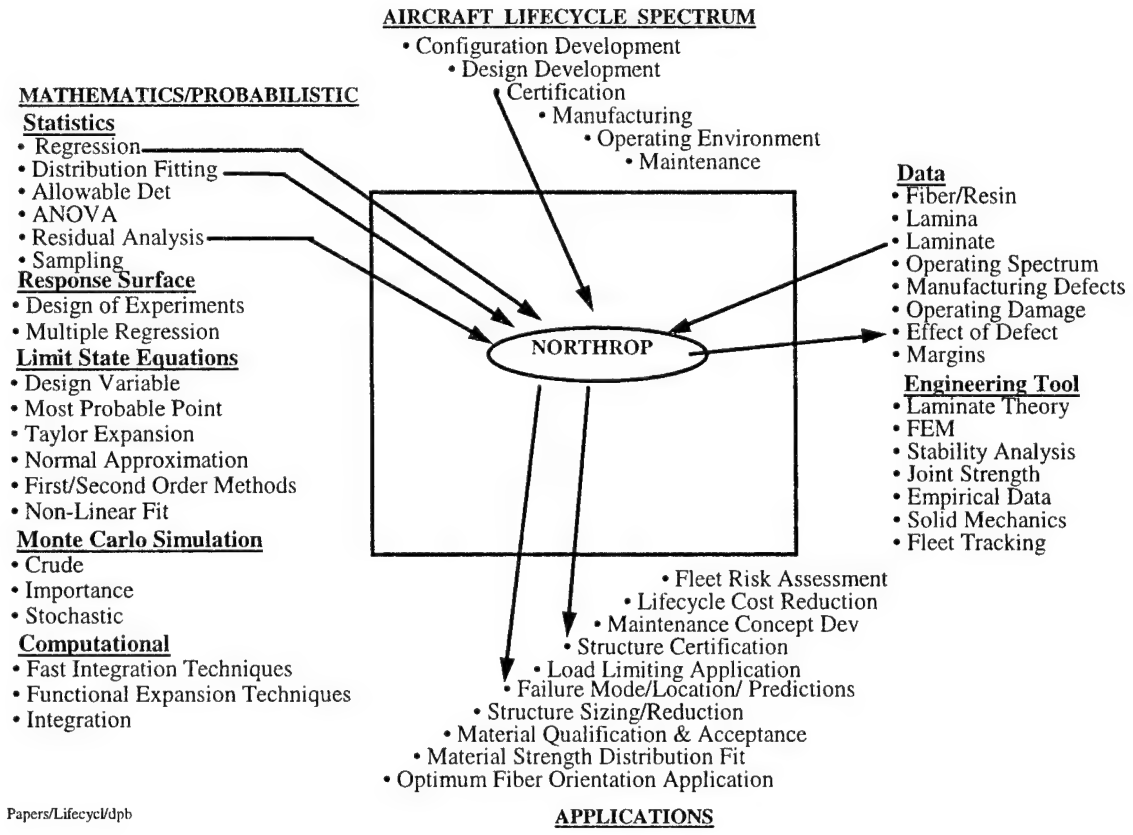


FIGURE 2-4. INDEPENDENT METHODOLOGY ASSESSMENT—NORTHROP APPROACH [2]

2.2.3 Probabilistic Maintenance Scheduling for Composite Aircraft [3].

Rouchon's work in composite probabilistic design deals with two major issues:

- Certification and compliance philosophy.
- Probabilistic inspection for fleet reliability.

2.2.3.1 Certification and Compliance Philosophy.

Rouchon's findings in the area of certification and compliance philosophy are based on experience from major programs starting with the carbon wing for the Falcon 10 V10F (for a description of this experimental version of the Falcon 10 see reference 4) to the latest Airbus aircraft and the Aerospatiale ATR 72. Specifically, three major issues were addressed: second source material qualification, conditions to simulate environmental effects, and damage tolerance demonstration for accidental impact damage.

The second source material qualification test is essentially the same as the t-test defined in MIL-HDBK-17 with the key material parameters subject to its pass/fail criterion.

2.2.3.2 Probabilistic Inspection Scheduling.

This issue is related to the issue of damage tolerance in connection with accidental service induced damage. Accidental damage is addressed in [3] through a scheduled inspection program based on the proportion of flight time variable. The proportion of flight time is defined as follows:

$$\text{Proportion of flight time} = \begin{array}{c}\text{The probability of failure per flight hour} \\ \text{X} \\ \text{The average time in failed condition}\end{array}$$

The average time in failed condition is assumed to equal 50 percent of the interval of time between inspections.

The methodology requires a comprehensive database on the probability of impact damage on structures allowing for the components involved, the aircraft operating conditions, location of damage on a given part, etc.

Rouchon recommended that the inspection program requires a probabilistic approach for its determination defined in reference 5. The probabilistic inspection concept is a new approach to certification which allows for inclusion of the maintenance philosophy at the design stage. The concept depends on a random sample size of aircraft to be chosen out of the fleet for inspection. The next inspection time is then defined based on the findings from current inspection. Total probability of failure will include not only the probability of structural failure but also the probability of failure due to structural damage which was not detected by the inspection program. Optimum definition of time between inspections and optimum life-cycle cost may be achieved by using this approach.

Rouchon's methodology makes assumptions about requirements on probabilistic and conditional probabilistic measures which include

- Allowed probability of structural failure is $\leq 1 \times 10^{-9}$ per flight hour.
- Probability of occurrence of a defined damage size is $\leq 1 \times 10^{-5}$ per flight hour.
- Probability of experiencing limit load and gust is $\leq 2 \times 10^{-5}$ per flight hour.
- Probability of experiencing ultimate load is $\leq 1 \times 10^{-8}$ per flight hour.

Rouchon illustrated the approach on inspection scheduling for the ATR 72 aircraft. The inspection interval is determined such that larger impact damage, where residual strength after impact is between limit and ultimate, corresponds to a calculation of the stress-strength failure probability which is less than 10^{-9} per flight hour. Low-level impact damage, which does not reduce strength below ultimate, is covered by a demonstration of no growth for the life of the aircraft. Figure 2-5 shows all inspection decisions as a function of the reduced strength after larger impact damage, length of inspection interval, and the corresponding probability of failure.

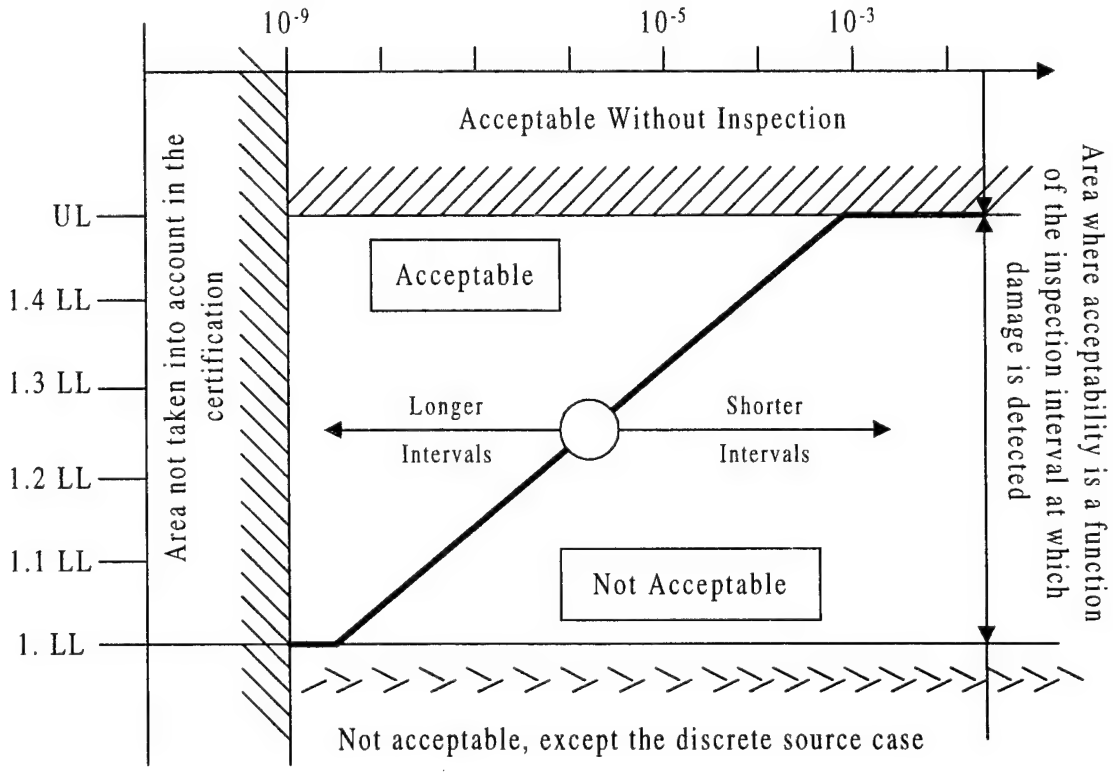


FIGURE 2-5. INSPECTION CRITERIA VERSUS LARGER IMPACT DAMAGE

Figure 2-6 summarizes all the input, output, and areas of application of the probabilistic maintenance scheduling approach as discussed by Rouchon [5].

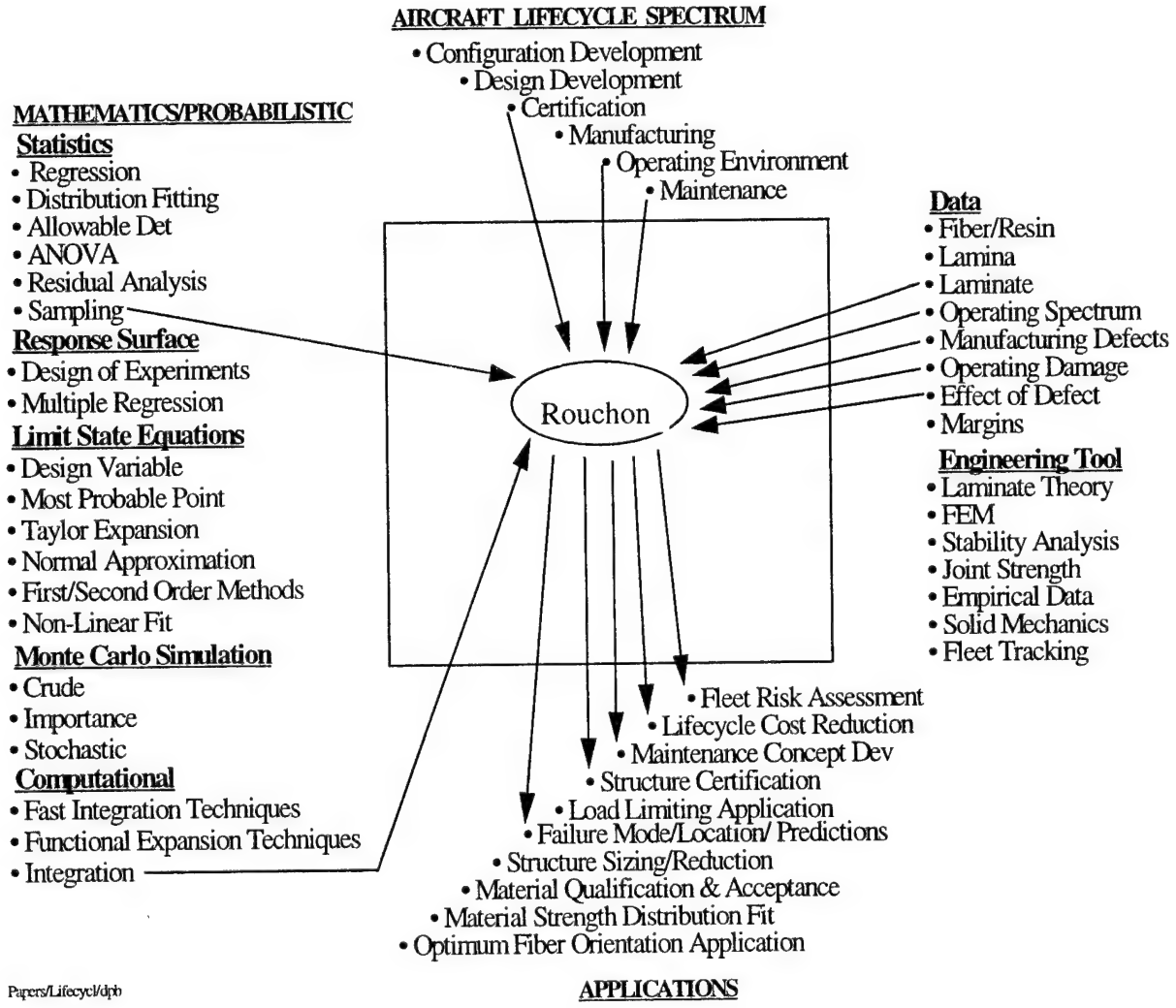


FIGURE 2-6. METHODOLOGY SUMMARY—ROUCHON[5]

3. NORTHROP GRUMMAN COMMERCIAL AIRCRAFT DIVISION (NGCAD) PROBABILISTIC DESIGN METHODOLOGY.

The NGCAD probabilistic design methodology, in development under IR&D since 1988, is a Monte Carlo simulation in which the distributions of operating stress and material strength are subjected to the lifetime risk drivers of material quality, manufacturing quality, thermal stress, gust, operating environment (moisture and temperature), and operational structural damage (hail, FOD, and maintenance induced damage). The distributions are adjusted to take into account the impact of all these random risk drivers. The probability of failure is calculated by integration as the probability of stress exceeding strength. Figure 3-1 shows the general derivation of this probability.

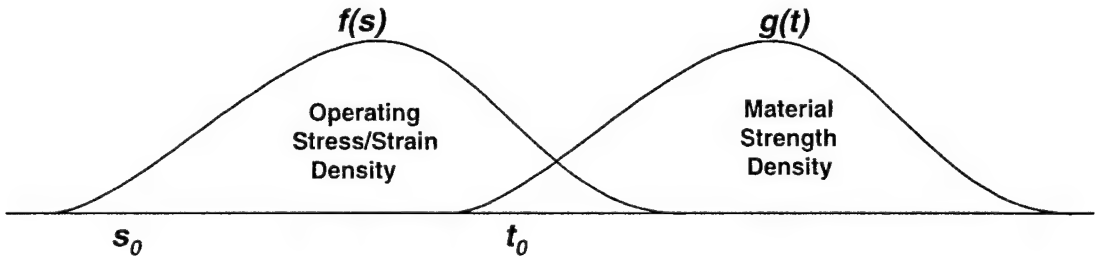


FIGURE 3-1. GENERAL DERIVATION OF PROBABILITY OF FAILURE

Note: In figure 3-1, material strength and operating stress/strain are assumed to be independent

- for S = stress/strain value, probability that material strength greater than S is

$$\int_s^\infty g(t)dt$$

- probability of component failure is

$$1-\int_{s_0}^{MAX[s_0,t_0]} f(s)ds - \int_{t_0}^{\infty} f(s)\left[\int_s^\infty g(t)dt\right]ds$$

Figure 3-2 shows a basic example of a probability of failure calculation.

From the figure, let both material strength and operating stress be defined by a normal distribution with

s= stress; with mean = 1000 psi and standard deviation = 100 psi
t= strength; with mean = 2340 psi and standard deviation = 200 psi

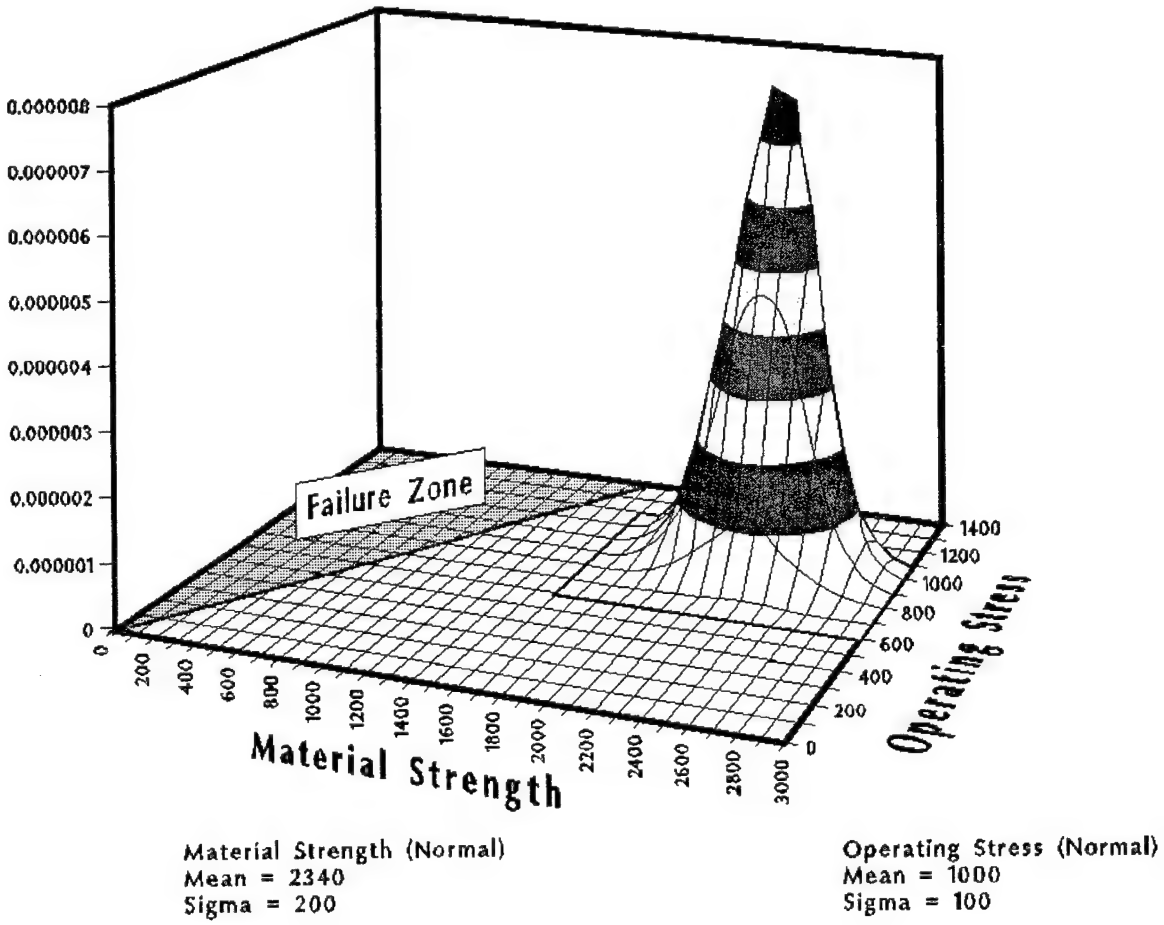


FIGURE 3-2. STRUCTURAL FAILURE PROBABILITY CALCULATION EXAMPLE

The applicable probability density functions (PDF's) are as follows:

$$\text{Stress:} \quad f(s) = \frac{1}{100\sqrt{2\Pi}} e^{-\frac{1}{2}\left(\frac{s-1000}{100}\right)^2}$$

$$\text{Strength:} \quad g(t) = \frac{1}{200\sqrt{2\Pi}} e^{-\frac{1}{2}\left(\frac{t-2340}{200}\right)^2}$$

Under the assumption of independence between material strength and operating stress, the joint PDF w(s,t) is

$$w(s,t) = \frac{1}{20{,}000(2\Pi)} e^{-\frac{1}{2}\left[\left(\frac{s-1000}{100}\right)^2 + \left(\frac{t-2340}{200}\right)^2\right]}$$

Figure 3-2 is a graphical representation of the surface w(s,t) for varying combinations of stress (s) and strength (t). The failure zone is defined as the area where stress exceeds strength and the probability of failure is the volume under the joint function w(s,t) corresponding to the failure zone.

Let $\Phi(z) = \frac{1}{\sqrt{2\Pi}} \int_{-\infty}^{z} e^{-\frac{1}{2}t^2} dt$ be the standard normal cumulative probability.

The failure probability is calculated as follows:

$$\text{Failure probability} = 1 - \frac{1}{200\sqrt{2\Pi}} \int_{-\infty}^{+\infty} e^{-\frac{1}{2}\left(\frac{t-2340}{200}\right)^2} \left[\frac{1}{100\sqrt{2\Pi}} \int_{t}^{+\infty} e^{-\frac{1}{2}\left(\frac{s-1000}{100}\right)^2} ds \right] dt$$

then

$$\text{Failure probability} = 1 - \frac{1}{200\sqrt{2\Pi}} \int_{-\infty}^{+\infty} e^{-\frac{1}{2}\left(\frac{t-2340}{200}\right)^2} \left[1 - \Phi\left(\frac{t-1000}{100}\right) \right] dt$$

$$= \frac{1}{200\sqrt{2\Pi}} \int_{-\infty}^{+\infty} e^{-\frac{1}{2}\left(\frac{t-2340}{200}\right)^2} \Phi\left(\frac{t-1000}{100}\right) dt$$

$$\text{let } r = \frac{t-2340}{200} \qquad \text{then} \qquad dr = \frac{1}{200} dt$$

$$\text{Failure probability} = \frac{1}{\sqrt{2\Pi}} \int_{-\infty}^{+\infty} e^{-\frac{1}{2}r^2} \Phi(2r + 13.40) dt = 1.5x10^{-9}$$

The distribution types for representation of stress or strain and material strength in this methodology are the normal, lognormal, and Weibull distributions. Other distributions used to fit operational parameters of the stress and strength PDF's are a modified triangular (referred to as the triflex distribution) and Poisson distributions. In the triflex distribution the triangular distribution is modified by the inclusion of a shape parameter to weight the data about its mean value. This is similar to the use of a shape parameter in the beta distribution to weight the data. The triflex also has the capability of emulating a uniform distribution.

Each component of the design is analyzed for all possible failure modes. For each failure mode, a corresponding distribution of material strength and operational stress is defined. Manufacturing defects and environmental and operational effects are accounted for in the simulation by modifications to the material strength distribution.

Standard spreadsheet and MIL-HDBK-17 [6] distributional fitting procedures are used on material property coupon data and on exceedance data. Integration is performed in quadruple precision as most of the answers are extremely small numbers.

Figure 3-3 shows the flowchart for the NGCAD baseline risk assessment procedure. Exceedance data is fitted to produce the load distribution. Material coupon data is analyzed and fitted to produce material strength distributions, material allowables, and material strength reductions due to moisture and temperature. Finite element data are analyzed to determine failure locations, failure modes, and margins of safety for the structure. For each location and failure mode, the simulation produces random manufacturing defects and random operating events such as hail, FOD, maintenance induced damage, random operating temperatures, and random moisture absorption levels (governed by the age of the structure).

During the Monte Carlo simulation, the following are randomly drawn:

- Flight Temperature (Discrete Distribution)
- Percent Moisture Absorbed (Triflex Distribution)
- Gust Occurrence (Discrete Distribution)
- Gust Magnitude (Uniform Distribution)
- Manufacturing Defects (Discrete Distribution)
- Operational Defects (Discrete Distribution)

3.1 REPRESENTATION OF FLIGHT TEMPERATURE.

A distribution of flight time at discrete temperatures (representing temperature intervals) is represented by a discrete distribution, shown in tabular form in section 4 (see table 4-9). The first temperature value, -65°F, has a discrete probability of 0.124; the structure temperature is assumed to be at -65°F 12.4 percent of the time. A cumulative discrete function is then defined, summing to 1.0 (100 percent of the time).

The Monte Carlo simulation involves choosing a random value from a uniform (from 0.0 to 1.0) distribution. The range bounding the random draw represents the structural temperature. For example, should the random number 0.10 be picked, falling in the range bounded by 0.0 and 0.1234, the structure temperature would be set to -65°F.

3.2 MOISTURE ABSORPTION REPRESENTATION.

Material strength is affected by moisture content as well as temperature. The Monte Carlo simulation chooses random percentages from a triflex distribution, bounded by 0 and 100 percent moisture material strength curves at the previously chosen temperature, as shown at the right side of figure 3-3. The randomly chosen percentage corresponds to a material strength reduction scale factor, based on a value of 1.0 at room temperature ambient conditions. These curves are discussed in paragraph 4.1 of section 4 for the Lear Fan aircraft.

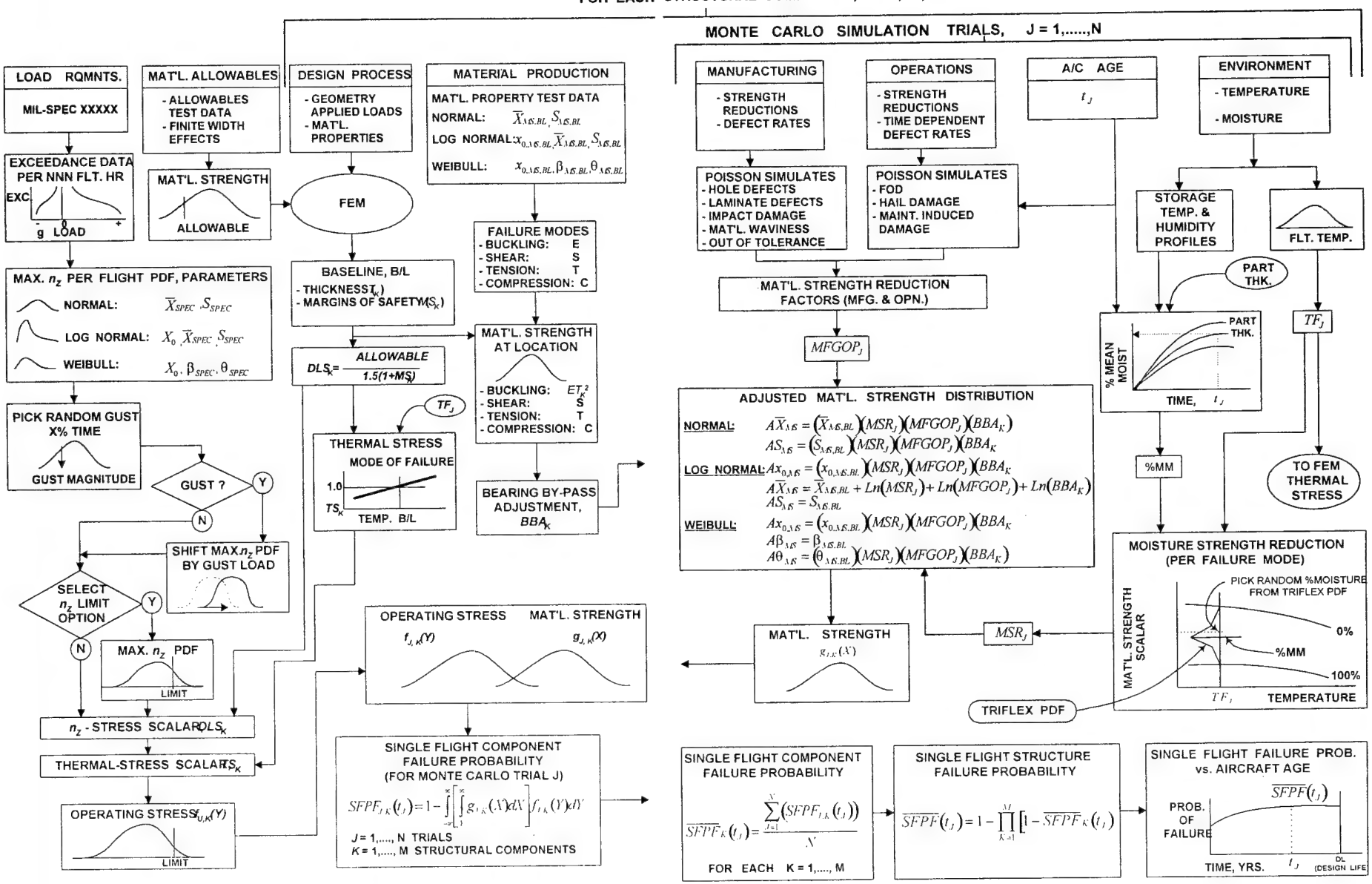


FIGURE 3-3. NGCAD PROBABILISTIC DESIGN MODEL

The effect of a scale factor on the three distributions is shown in figure 3-4. A scale factor is a multiplicative factor, as opposed to a translational shift, applied to each distributional value. In the normal distribution case, the scale factor multiplies both mean and standard deviation in order to maintain a constant coefficient of variation.

$$\text{Normal: } g(t) = \frac{1}{\sqrt{2\pi}\sigma_t} e^{-\frac{1}{2}\left(\frac{t-\mu_t}{\sigma_t}\right)^2}$$

$$\text{Lognormal: } g(t) = \frac{1}{(t-t_0)\sqrt{2\pi}\sigma_t} e^{-\frac{1}{2}\left(\frac{\ln(t-t_0)-\mu_t}{\sigma_t}\right)^2}$$

$$\text{Weibull: } g(t) = \frac{\beta_t}{\sigma_t}\left(\frac{t-t_0}{\sigma_t}\right)^{\beta_t - 1} e^{-\left(\frac{t-t_0}{\sigma_t}\right)^{\beta_t}}$$

Distribution	Parameters	Transformation	New Parameters
Normal	μ_t, σ_t	Shift C_1	$\mu_t + C_1$, σ_t
Lognormal	t_0, μ_t, σ_t	Shift C_1	$t_0 + C_1$, μ_t, σ_t
Weibull	t_0, β_t, σ_t	Shift C_1	$t_0 + C_1$, β_t, σ_t
Normal	μ_t, σ_t	Scale C_2	$C_2\ \mu_t$, $\lvert C_2 \rvert \sigma_t$
Lognormal	t_0, μ_t, σ_t	Scale C_2	$C_2\ t_0$, $\mu_t + \ln \lvert C_2 \rvert$, σ_t Reverse Domain for $C_2 < o$
Weibull	t_0, β_t, σ_t	Scale C_2	$C_2\ t_0$, β_t, $\lvert C_2 \rvert \sigma_t$ Reverse Domain for $C_2 < o$

FIGURE 3-4. DISTRIBUTION FUNCTIONAL FORMS AND TRANSFORMATIONS

3.3 GUST LOADING.

Gust loading is implemented as an event that happens a portion of the lifetime. From the input of probability of gust occurring (for Lear Fan: 0.1), a discrete distribution is defined to delineate between a gust and no gust situation. During the Monte Carlo simulation, a random value is drawn from a uniform (0.0 to 1.0) distribution. The range bounding the random draw represents the occurrence (or nonoccurrence) of gust. That is, random picks from 0.0 to 0.1 would indicate occurrence of gust, while values above 0.1 up to 1.0 indicate no gust.

Should a gust occur, the Monte Carlo simulation chooses a random gust load factor from a uniform distribution. The value chosen will shift (translate) the load factor distribution to the left or right but will not alter the distribution shape. Figure 3-4 shows how the shift transformations are handled for each distribution type.

3.4 MANUFACTURING AND OPERATIONAL DEFECTS.

Discrete occurrence probabilities (from 0.0 to 1.0) for each type of manufacturing and operational defect are calculated from input defect rates. These rates, based on NGCAD experience (manufacturing) and from commercial operations failure data analysis, are given as expected undetected defects per square foot for manufacturing defects and expected number of defects per square foot per flight hour for operational defects. These defect rates are listed in appendix C, pages C-6 and C-7. Coupled with the location areas and analysis time, discrete probabilities can be determined for each location by use of the Poisson distribution.

The discrete probability for one or more defects occurring would be one minus the probability that no defects occur. Mathematically,

$$P(\text{1 or more defects}) = 1.0 - P(\text{no defects})$$

where

$$P(\text{no defects}) = \text{EXP}(-1 \times \text{defect rate} \times \text{area})$$

For example, if the manufacturing defect rate for waviness is 0.0197 and the location area of interest is 2 square feet, then

$$P(\text{1 or more defects}) = 1.0 - \text{EXP}(-0.0197 \times 2.0)$$
$$= 0.0386$$

During the Monte Carlo simulation, any random pick from 0.0 to 0.0386 would yield a waviness defect, and the appropriate material strength parameter would be scaled in accordance with figure 3-4.

If, for that same location, the operational defect rate was $1.0E^{-6}$ per square foot per flight hour, the analysis point was 15,000 flight hours (FH), and the location area was 2.0 square feet, the discrete probability for that defect would be

$$P(\text{1 or more defects}) = 1.0 - \text{EXP}(-1 \times 1.0E\text{-}6 \times 15{,}000 \times 2.0)$$
$$= 0.0296$$

During the Monte Carlo simulation, any random pick from 0.0 to 0.0296 would yield an operational defect, and the appropriate material strength parameter would be scaled in accordance with figure 3-4.

The impact of these random events on the stress or strain distribution as well as the material strength distribution is accounted for by proper scaling and shifting per figure 3-3. The probability of structural failure and the failure mode at a given location are then found by the integration, as indicated in figure 3-1.

Probability of failure for various locations and failure modes is used to determine the single-flight hour probability of failure of the structure. This is also used to determine the expected failures for a fleet of aircraft.

The choice of the Monte Carlo simulation approach allows for the use of any type of distribution for material strength and operating stress/strain. This, in essence, allows the treatment of the goodness of fit of the statistical database to be done independently from the structure risk assessment. Additionally, risk drivers and variables may be added or taken out of the simulation without requiring major modification to the model. Figure 3-5 provides a summary of all input, output, and areas of application of the NGCAD probabilistic design methodology.

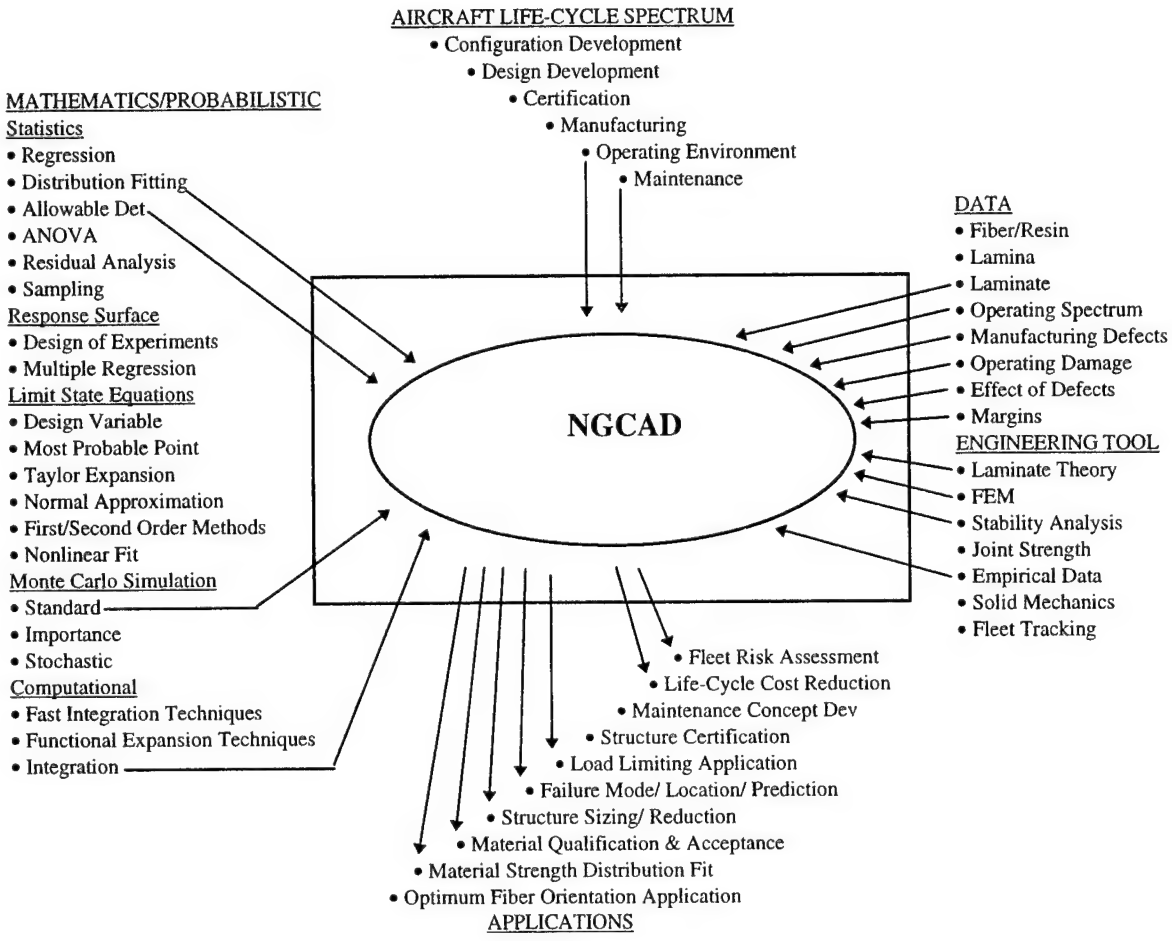


FIGURE 3-5. NGCAD PROBABILISTIC DESIGN METHODOLOGY SUMMARY

4. PROBABILISTIC RISK ANALYSIS OF THE LEAR FAN COMPOSITE WING.

Lear Fan reports and documents listed in references 7-13 were made available by the FAA; except where noted below, data required to perform the NGCAD probabilistic analysis on the wing box was found in these references. Key data items include

- Static test strain measurements (over 1000 μ in/in) for the upper wing skin, lower wing skin, and substructure locations.
- Material strength characterization for failure modes of interest (material allowables, coupon data).
- Spectra loading (mission profiles).
- Flight temperatures (time at temperatures).

Manufacturing defect and operational damage rates were generated using NGCAD's experience and information from recent visits to airline maintenance facilities.

4.1 MATERIAL ALLOWABLES (COUPON DATA).

References 5 and 6 provided material property data for this study. Table 4-1 lists the materials described in these reports, and shows the initial test data (denoted ORIG) and additional data (denoted ADDED) which was obtained two years later. The Lear Fan reports provided coupon-level results for approximately 15 percent of the LMS 1200 unidirectional tape tension and compression test specimens. For the LMS 1201 bi-directional fabric, nearly 60 percent of the tension and compression coupon-level results were provided. The initial data provided individual coupon test results, whereas the additional data provided only summary statistics including mean, sigma, sample size, and B-basis value. Therefore, the ADDED data could not be statistically fit for lack of individual coupon values. For analysis purposes, this added data was assumed to be normally distributed.

Material Descriptions:

LMS 1200	→	Unidirectional Tape: Fiberite Type HY-E-1034C
LMS 1201	→	Bidirectional Fabric: Fiberite Type HMF 133/34

To perform the statistical analysis to produce material allowable distributions and allowable data, coupon data is customarily analyzed using the MIL-HDBK-17 [6] STAT17 program. However, since the data was mostly given in summary statistics, the STAT17 analysis was not done in this case and the material allowable distributions are assumed to be normal as it was reported in the test results interim report [7]. Table 4-2 shows mechanical property distribution parameters.

TABLE 4-1. LEAR FAN MATERIAL PROPERTY DATA

MATERIAL TYPE	DIRECTION	PROPERTY	TEMPERATURE	NUMBER		INDIVIDUAL TEST DATA
LMS 1200	0 DEGREES	TS, TM	-65°F	24	(Orig)	Yes
				88	(Added)	No
			RT	12	(Orig)	Yes
				94	(Added)	No
			180°F	15	(Orig)	Yes
		CS, CM	-65°F	8	(Orig)	Yes
				72	(Added)	No
			RT	0	(Orig)	---
				71	(Added)	No
			180°F	25	(Orig)	Yes
				64	(Added)	No
		SHEAR	-65°F	35	(Orig)	Yes
				114	(Added)	No
			RT	35	(Orig)	Yes
				113	(Added)	No
			180°F	34	(Orig))	Yes
				108	(Added)	No
LMS 1201	WARP	TS, TM	-65°F	100	(Orig)	Yes
				35	(Added)	No
			RT	89	(Orig)	Yes
				116	(Added)	No
			180°F	93	(Orig)	Yes
				34	(Added)	No
		CS, CM	-65°F	88	(Orig)	Yes
				35	(Added)	No
			RT	92	(Orig)	Yes
				104	(Added)	No
			180°F	93	(Orig)	Yes
				71	(Added)	No
		SHEAR	-65°F	95	(Orig)	Yes
				144	(Added)	
			RT	95	(Orig)	Yes
				145	(Added)	No
			180°F	95	(Orig)	Yes
				140	(Added)	No
LMS 1201	FILL	TS, TM	-65°F	104	(Orig)	Yes
				35	(Added)	No
			RT	88	(Orig)	Yes
				117	(Added)	No
			180°F	95	(Orig)	Yes
				34	(Added)	No
		CS, CM	-65°F	94	(Orig)	Yes
				31	(Added)	No
			RT	88	(Orig)	Yes
				65	(Added)	No
			180°F	95	(Orig)	Yes
				35	(Added)	No

TABLE 4-2. SUMMARY OF LMS 1200 AND 1201 MECHANICAL PROPERTY DISTRIBUTION PARAMETERS (NORMALLY DISTRIBUTED)

MATERIAL TYPE	PROPERTY	TEMP	NUMBER	MEAN	STD DEV
LMS 1200 (0 Degree)	Tension Strength	-65°F	112	9930 μ in/in	1336 μ in/in
		Room Temp	106	10550 μ in/in	1336 μ in/in*
		180°F	105	10540 μ in/in	1336 μ in/in*
	Compression Strength	-65°F	80	13080 μ in/in	1763 μ in/in
		Room Temp	71	11560 μ in/in	1368 μ in/in
		180°F	89	10940 μ in/in	1587 μ in/in
	Compression Modulus	-65°F	80	18.7 Msi	0.94 Msi**
		Room Temp	71	18.8 Msi	0.94 Msi**
		180°F	89	18.4 Msi	0.92 Msi**
LMS 1201 (Warp)	Tension Strength	-65°F	135	9530 μ in/in	738 μ in/in
		Room Temp	205	8900 μ in/in	836 μ in/in
		180°F	127	9040 μ in/in	574 μ in/in
	Compression Strength	-65°F	123	10780 μ in/in	1612 μ in/in
		Room Temp	196	9940 μ in/in	1315 μ in/in
		180°F	164	9040 μ in/in	1273 μ in/in

* Data illegible: used same value as at -65 ° F

** Data unavailable: value calculated by assuming COV of 5% (representative of AS-4/3502 compression modulus COV)

The effect of temperature on material properties was obtained from [8]. Testing was done at -65°F, room temperature, and 180°F; all tests were conducted using "dry" specimens, which typically contain approximately 0.2 percent moisture. The compression strength knockdown factors due to moisture absorption were obtained from the NGCAD database on the AS-4/3502 graphite/epoxy material.

Figures 4-1 through 4-5 show the resulting knockdown factors, based on a room temperature dry baseline value, for the applicable LMS 1200 and LMS 1201 failure modes.

The lower wing skin drawings were unavailable; these would have indicated ply stacking sequences for the various critical locations. However, since the lower skin critical locations are in the built-up areas of the spar attachment regions, it was assumed that the bending load is primarily reacted by 0° tape plies. This assumption leads to using 0° tape tension coupon data for the material strength distribution on the lower skin.

Similarly, no substructure drawings were available. After review of typical composite shear web designs, it was assumed the ply stackup was primarily ±45° plies, and therefore the LMS 1201 fabric compression and tension coupon data was used for the material strength distribution on the substructure.

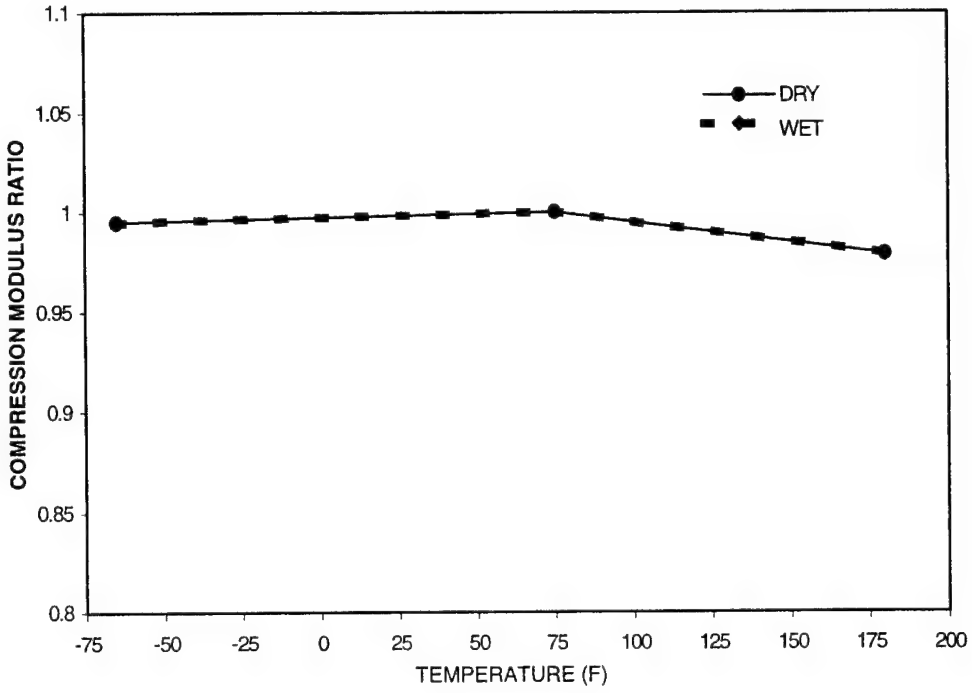


FIGURE 4-1. EFFECT OF ENVIRONMENT ON LMS 1200 TAPE COMPRESSION MODULUS

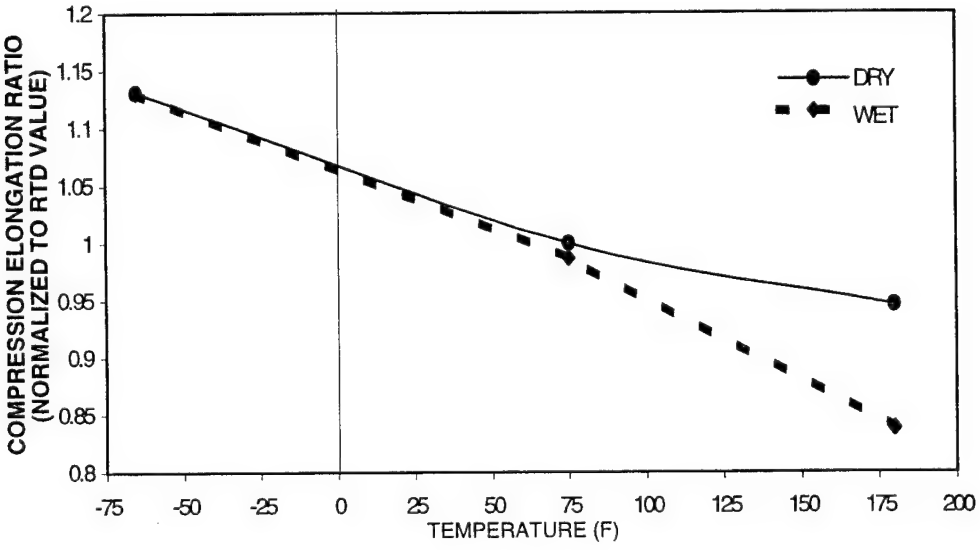


FIGURE 4-2. EFFECT OF ENVIRONMENT ON LMS 1200 TAPE COMPRESSION ELONGATION

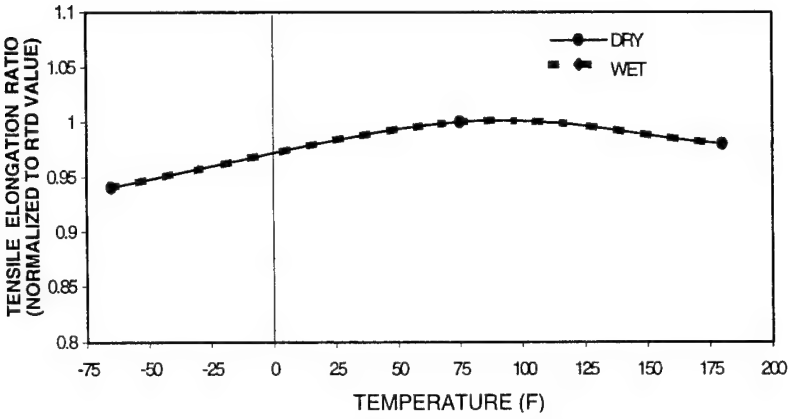


FIGURE 4-3. EFFECT OF ENVIRONMENT ON LMS 1200 TAPE TENSION ELONGATION

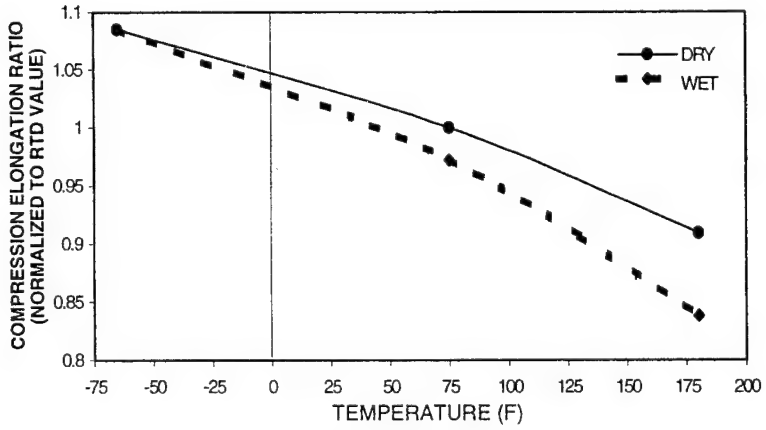


FIGURE 4-4. EFFECT OF ENVIRONMENT ON LMS 1201 FABRIC COMPRESSION ELONGATION

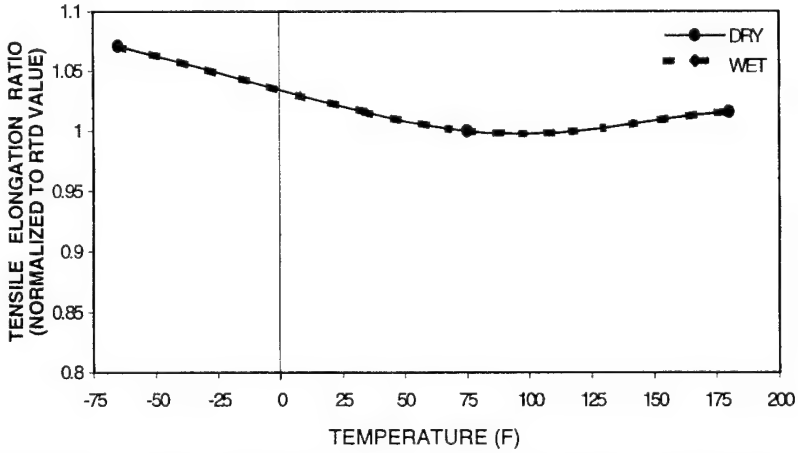
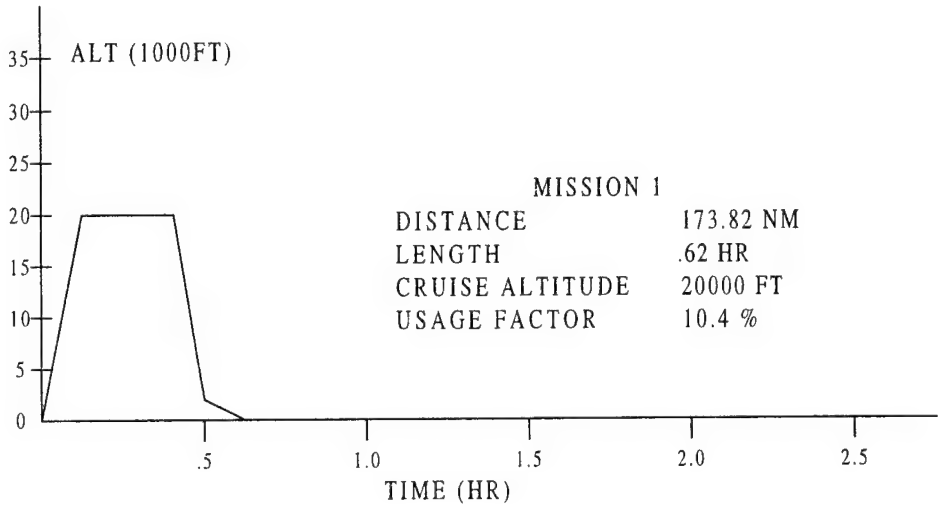


FIGURE 4-5. EFFECT OF ENVIRONMENT ON LMS 1201 FABRIC TENSION ELONGATION

4.2 SPECTRA LOADING (MISSION PROFILES).

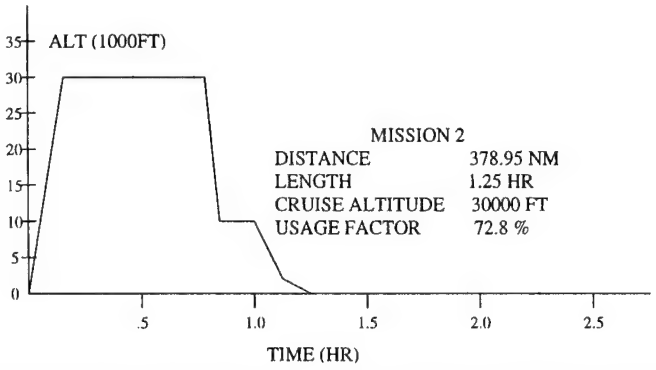
The development of the Lear Fan mission spectra required defining the manner in which the aircraft would be used over its lifetime. Three missions were originally selected to represent the predicted usage of the Lear Fan 2100. These mission profiles, presented in figures 4-6 through 4-8, were combined as discussed below to form the composite spectrum presented in figure 4-9 that defined the flight regime in terms of symmetric flight maneuver experience. Gust data including up gust and down gust, were analyzed and fitted statistically as discussed in section 3. Gust loading was accounted for as shown in the NGCAD methodology flow chart of figure 3-3.



SEGMENT	TYPE	MEAN ALTITUDE	VELOCITY (KEAS*)	MEAN WEIGHT	TIME (MIN)	MISSION FACTOR TIME (%)
1	Climb	2500	175	7346.	1.4	3.78
2	Climb	7500	175	7338.	1.5	4.05
3	Climb	12500	175	7330.	1.6	4.32
4	Climb	17500	175	7322.	1.7	4.59
5	Cruise	20000	270	7268.	9.5	25.68
6	Cruise	20000	270	7169.	9.5	25.68
7	Descent	6000	240	7096.	6.0	16.22
8	Approach	1000	90	7057.	6.0	16.22

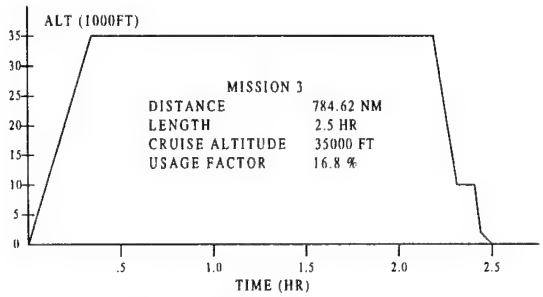
*KEAS—Knots Equivalent Airspeed

FIGURE 4-6. LEAR FAN MISSION 1 SEGMENTS



SEGMENT	TYPE	MEAN ALTITUDE	VELOCITY (KEAS)	MEAN WEIGHT	TIME (MIN)	MISSION FACTOR TIME (%)
1	Climb	2500	175	7346	1.15	1.53
2	Climb	7500	175	7339	1.3	1.73
3	Climb	12500	175	7332	1.45	1.93
4	Climb	17500	175	7325	1.9	2.54
5	Climb	22500	175	7315	2.35	3.14
6	Climb	27500	175	7303	3.3	4.40
7	Cruise	30000	220	7205	21.15	28.22
8	Cruise	30000	220	7024	21.15	28.22
9	Descent	16000	225	6913	6.2	8.27
10	Cruise	10000	250	6870	6.0	8.01
11	Descent	6000	230	6835	3.0	4.00
12	Approach	1000	90	6809	6.0	8.01

FIGURE 4-7. LEAR FAN MISSION 2 SEGMENTS



SEGMENT	TYPE	MEAN ALTITUDE	VELOCITY (KEAS)	MEAN WEIGHT	TIME (MIN)	MISSION FACTOR TIME (%)
1	Climb	2500	175	7347	1.15	.77
2	Climb	7500	175	7339	1.3	.87
3	Climb	12500	175	7332	1.45	.97
4	Climb	17500	175	7325	1.9	1.27
5	Climb	22500	175	7315	2.35	1.57
6	Climb	27500	173	7303	3.3	2.21
7	Climb	32500	160	7287	4.6	3.08
8	Cruise	35000	190	7110	55.0	36.79
9	Cruise	35000	190	6773	55.0	36.79
10	Descent	22500	215	6581	8.25	5.52
11	Cruise	10000	250	6533	6.0	4.01
12	Descent	6000	230	6498	3.0	2.01
13	Approach	1000	90	6471	6.0	4.01

FIGURE 4-8. LEAR FAN MISSION 3 SEGMENTS

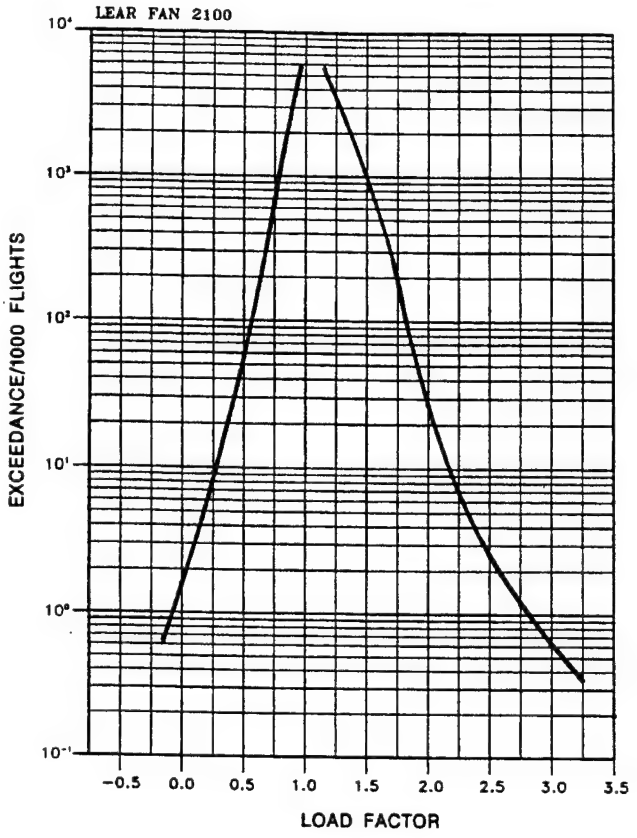


FIGURE 4-9. LEAR FAN COMBINED EXCEEDANCES CURVE

The NGCAD methodology described in figure 3-3 used the Lear Fan maneuver exceedance data to create an aircraft lifetime maneuver occurrences spectrum. A Poisson distribution was then used to create an n_Z max per Δt distribution. The parameter λ was defined as the expected number of times the load factor n_{Z_i} will be exceeded during the time interval Δt. Assuming a Poisson distribution, $e^{-\lambda_i}$, thus represents the probability that n_{Z_i} will not be exceeded, hence the probability that the maximum n_Z will lie between 1.0 g and n_{Z_i} during Δt. Therefore, $e^{-\lambda_i}$ is the value of n_Z max per Δt cumulative distribution function (CDF) at n_{Z_i}, or equivalently, the area under the n_Z max per Δt probability density function (PDF) between 1.0 g and n_{Z_i}. This process is repeated for as many n_{Z_i} values as necessary to adequately define the n_Z max PDF for interval Δt. Statistical analyses of the occurrences spectrum, using the MIL-HDBK-17 methodology (section 8.6.4 of [6]), indicated the best curve fit would be a 3 parameter, lognormal distribution with $Z_o = 1.00$, $\mu = -1.06645$, and $\sigma = 0.53677$. This probability density function is shown in figure 4-10, and is used at all the analysis points in the NGCAD probabilistic methodology. However, the code is capable of accommodating individual probability density functions at each analysis point, if necessary. The original Lear Fan 2100 life was defined as 15,000 hours with 12,274 flights. The maximum load factor was 3.5 g's with a maximum takeoff weight of 7,350 lbs.

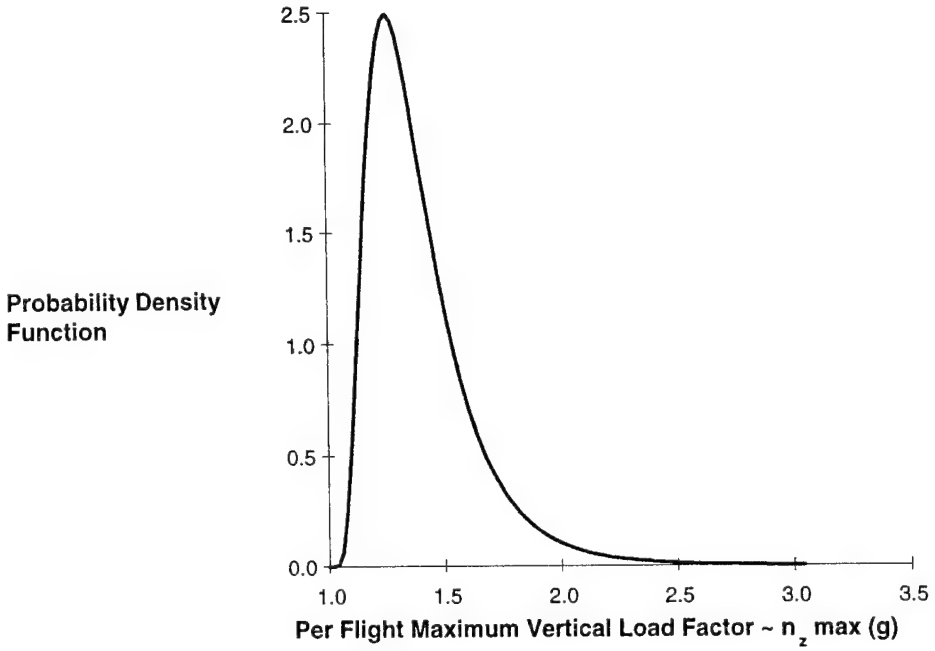


FIGURE 4-10. LEAR FAN G'S PROBABILITY DENSITY FUNCTION

Utilization factors for the three missions were given as 10.4, 72.8, and 16.8 percent. Since the total lifetime flight hours is given as 15,000 hours, then the corresponding flight hours for the three missions are calculated by

Mission #1 total flight hours = 15000 x 10.4% = 1,560 hours
Mission #2 total flight hours = 15000 x 72.8% = 10,920 hours
Mission #3 total flight hours = 15000 x 16.8% = 2,520 hours

The length of flight for missions 1, 2, and 3 were given as 0.62, 1.25, and 2.5 hours, respectively.

The following data shows the remainder of the calculations leading to the number of flights for each mission.

Mission	Average Flight Hours Per Flight	Total Flight Hours	Total Number Flights
1	0.62	1560	2516
2	1.25	10920	8736
3	2.5	2520	1008

This calculation produces a total number of flights of 12,260 which is very close to the 12,274 flights, corresponding to the design life. The difference is attributed to the roundoff of the average mission flight hours.

4.3 APPLIED LOADINGS AND CRITICAL AREAS.

The internal strains, resulting from externally applied loads, and critical locations were obtained from the Lear Fan 2100 Wing Static Test Results [11].

The test condition selected for the NGCAD Probabilistic Analysis was the symmetric up bending case which is consistent with the selected maneuver spectrum. Limit loads were increased by the static test factor of 1.14 to obtain the adjusted limit test load factor used to satisfy the Federal Aviation Administration (FAA) limit load requirement [15, 16] that loads to be used for static testing of dry structures at room temperature be increased to account for the effects of material variability and environment as follows:

Adjusted Test Load = FAR 23 Test Load x Static Test Load Factor

where

Static Test Load Factor = Variability Factor x Environmental Factor

The material variability factor ("scatter" factor) is defined as typical room temperature (dry) material strength divided by its B-basis allowable strength, while the environmental factor is defined as the room temperature (dry) typical strength divided by its typical strength after worst case environmental conditioning. The Lear Fan variability factor was 1.056 [10, p. 2.3] while the environmental factor was 1.08 [10, p. 1.3]. Therefore, the static test load factor was calculated as follows:

Static Test Load Factor = 1.056 x 1.08 = 1.14

and therefore

Adjusted Test Load = FAR 23 Test Load x 1.14

FAA-adjusted ultimate test load requirements were satisfied by further increasing the adjusted limit test loads by a factor of 1.5. Therefore, design ultimate load is given by 1.5 x design limit load x 1.14. During loading to symmetric up bending ultimate load, skin buckles were observed outboard of Y 49 and loading was discontinued at 94 percent of the 275 knots equivalent airspeed (KEAS) flight envelope ultimate load. The flight envelope was reduced accordingly.

The Lear Fan 2100 Wing Static Test Results report [11] defined critical strains as those exceeding 2,000 μ in/in. Critical symmetric up bending test strains at selected strain gage locations at 94 percent ultimate load are presented in tables 4-3 and 4-4. The indicated ID points are the critical points (denoted as high-strain locations) selected. Strain gage locations are shown in figures 4-11 through 4-13.

TABLE 4-3. CRITICAL STATIC TEST STRAIN MEASUREMENTS

Gage Location and Axis	Micro-strain	Load Condition	Channel Number	ID Points
Upper wing skin, center spar datum, Yw 38.5, axis Ycs	-2420	Maximum positive torque, ultimate load	8	
Lower wing skin, center spar datum, Yw 38.5, axis Ycs	2010	Maximum positive torque, ultimate load	15	
Upper wing skin, center spar datum, Yw 38.5, axis Ycs	-3390	Symmetric up bending, ultimate load	8	1
Upper wing skin front spar datum, Yw 38.5, axis Ycs	-2070	Symmetric up bending, ultimate load	9	
Upper wing skin Xcs = -7.9, Yw 38.5 axis Ycs	-2440	Symmetric up bending, ultimate load	11	
Upper wing skin Xcs = -7.9, Yw 38.5 axis Ycs	-2670	Symmetric up bending, ultimate load	12	
Upper wing skin Xcs = 12.0. Yw 34.7 axis Ycs	-2920	Symmetric up bending, ultimate load	13	
Upper wing skin Xcs = 12.0, Yw 34.7, axis Ycs	-3669	Symmetric up bending, ultimate load	14	
Lower wing skin, center spar datum, Yw 38.5, axis Ycs	2700	Symmetric up bending, ultimate load	15	22
Lower wing skin, front spar datum, Yw 38.5, axis Ycs	2270	Symmetric up bending, ultimate load	16	23
Lower wing skin, rear spar datum, Yw 38.5, axis Ycs	2420	Symmetric up bending, ultimate load	17	24
Rear spar web, midheight, Yw 38.5, axis Ycs	-3010	Symmetric up bending, ultimate load	22	44
Rear spar web, midheight, Yw 38.5, axis Ycs	2120	Symmetric up bending, ultimate load	23	45
Rib at Yw 28, Xcs = 15.25, Zrp = 2.0, axis Yrp	-2050	Symmetric up bending, ultimate load	24	46
Upper wing skin, center spar datum, Yw 78.9, axis Ycs	-2820	Symmetric up bending, ultimate load	27	3
Upper wing skin, center spar datum, Yw -38.5, axis Ycs	-2240	Symmetric up bending, ultimate load	60	
Upper wing skin, center spar datum, Yw 118, axis Ycs	-2950	Symmetric up bending, ultimate load	27	4
Lower wing skin, center spar datum, Yw 83.1, axis Ycs	2370	Symmetric up bending, ultimate load	43	26
Lower wing skin center spar datum, Yw 118.6, axis Ycs	2770	Symmetric up bending, ultimate load	44	28
Lower wing skin, rear spar datum, Yw 118, axis Ycs	2260	Symmetric up bending, ultimate load	48	29

TABLE 4-3. CRITICAL STATIC TEST STRAIN MEASUREMENTS (CONTINUED)

Gage Location and Axis	Micro-strain	Load Condition	Channel Number	ID Points
Lower wing skin, Xcs = 18.5, Yw 72.8, axis Ycs	2000	Symmetric up bending, ultimate load	54	27
Rib at Yw 88, Zrp = 0.5, Xcs 8.0, axis Ycs	-2040	Symmetric up bending, ultimate load	88	47
Upper wing skin, center spar datum, Yw 38.5, axial	-2200	E004 wing symmetric up bending with pressure	83	
Upper wing skin, front spar datum, Yw 38.5, axial	-1460	E004 wing symmetric up bending with pressure	85	
Upper wing skin, center spar datum, Yw 118, axial	-2060	E004 wing symmetric up bending with pressure	84	
Upper wing skin, OML, Yw 56.4, axial	-2170	E004 wing symmetric up bending with pressure	88	
Upper wing skin, IML, Yw 56.4, axial	-2000	E004 wing symmetric up bending with pressure	80	

Gage location references:

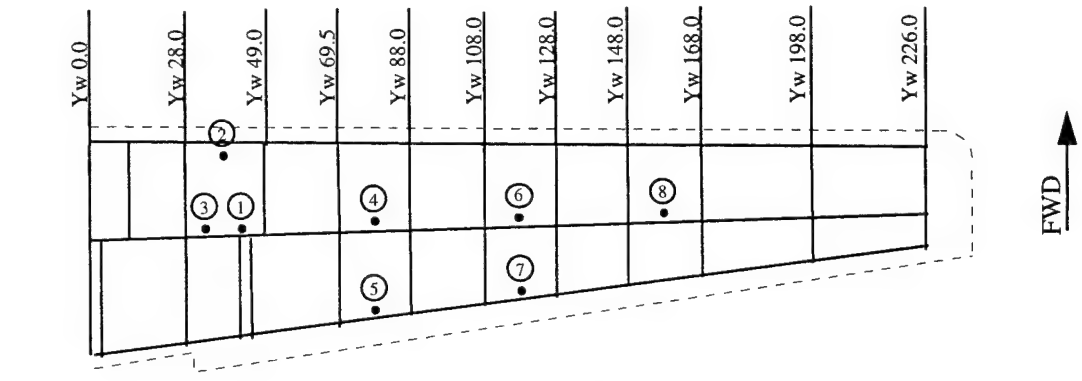
Xcs —referenced to center spar datum; positive aft.
Yw —referenced to aircraft centerline: negative left.
Zrp —referenced to wing reference plane: positive up.

Axis direction legend:

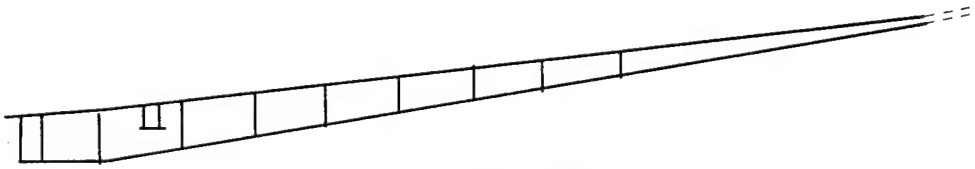
Xcs —perpendicular to center spar datum
Ycs —parallel to center spar datum.
/cs —45 degrees to center spar datum and 90 degrees to \cs.
\cs —45 degrees to center spar datum and 90 degrees to /cs.
Xrp —parallel to wing reference plane.
Zrp —perpendicular to wing reference plane.
/rp —45 degrees to wing ref plane and 90 degrees to \rp.
\rp —45 degrees to wing ref plane and 90 degrees to /rp.
/si —45 degrees to spar center line and 90 degrees to \si.
\si —45 degrees to spar center line and 90 degrees to /si.

TABLE 4-4. SYMMETRIC UP BENDING REDUCED STRAIN DATA
(see table 4-3 for notation)

Gage Location and Axis	Load = 33% of Ult.	Load = 67% of Ult.	Residual	Load = 94% of Ult.	Channel Number	ID Points
Upper wing skin, center spar datum, Yw 78.9, axis Ycs	-940	-2000	-10	---	27	
Upper wing skin, center spar datum, Yw 158, axis Ycs	-660	-1420	0	---	28	5
Upper wing skin, front spar datum, Yw 78.9, axis Ycs	-640	-1370	0	---	29	
Upper wing skin, rear spar datum, Yw 78.9, axis Ycs	-500	-1080	0	---	31	
Upper wing skin, Xcs = 10.4, Yw 78.9, axis \cs	-500	-1070	0	---	36	
Lower wing skin, center spar datum, Yw 78.8, axis Ycs	810	1720	10	---	41	25
Lower wing skin, center spar datum, Yw 78.8, axis Ycs	590	1260	10	---	42	
Lower wing skin, front spar datum, Yw 78.8, axis Ycs	530	1120	0	---	43	
Lower wing skin, rear spar datum, Yw 78.7, axis Ycs	580	1290	-40	---	45	
Lower wing skin, Xcs = 9.3, Yw 88, axis Xcs	-660	-1280	-10	---	55	
Upper wing skin, center spar datum, Yw -78.5, axis Xcs	-1050	-2240	0	---	60	
Upper wing skin, front spar datum, Yw -38.5, axis Ycs	-730	-1560	0	---	62	2
Upper wing skin, Xcs = 6.6, Yw -158.2, axis /cs	-200	-510	-140	---	70	
Lower wing skin, center spar datum, Yw -158, axis Ycs	630	1340	0	---	73	
Lower wing skin, rear spar datum, Yw -38.5, axis Ycs	780	1640	20	---	75	
Lower wing skin, Xcs = -5.5, Yw -158, axis \cs	550	1160	0	---	79	
Lower wing skin, center spar datum, Yw -38.5, axis Ycs	870	1840	0	---	80	
Lower wing skin, front spar datum, Yw -38.5, axis Ycs	650	1400	0	---	81	
Upper wing skin, center spar datum, Yw 118, axis Ycs	-95	-2000	---	-2950	27*	
Upper wing skin, front spar datum, Yw 118, axis Ycs	-630	-1310	---	-1890	29*	
Upper wing skin, rear spar datum, Yw 118, axis ycs	-570	-1190	---	-1890	31*	
Upper wing skin, Xcs = -6.2, Yw 118, axis \cs	-480	-1030	---	-1750	34*	
Lower wing skin, center spar datum, Yw 83.1, axis Ycs	770	1620	---	2370	43*	



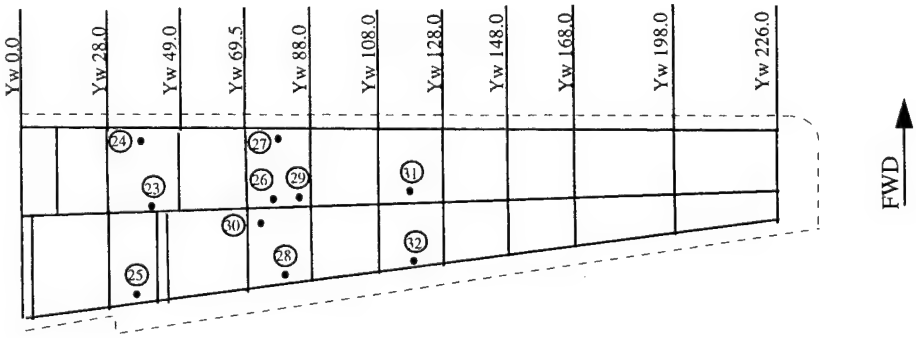
TOP VIEW



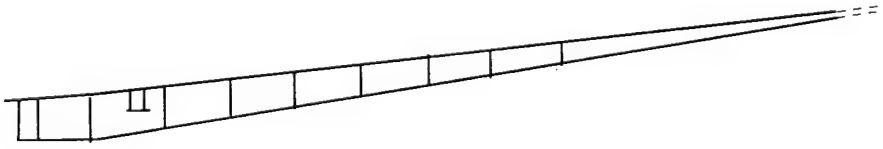
SIDE VIEW

LOOKING FORWARD

FIGURE 4-11. UPPER SKIN ANALYSIS LOCATIONS



BOTTOM VIEW



SIDE VIEW

LOOKING FORWARD

FIGURE 4-12. LOWER SKIN ANALYSIS LOCATIONS

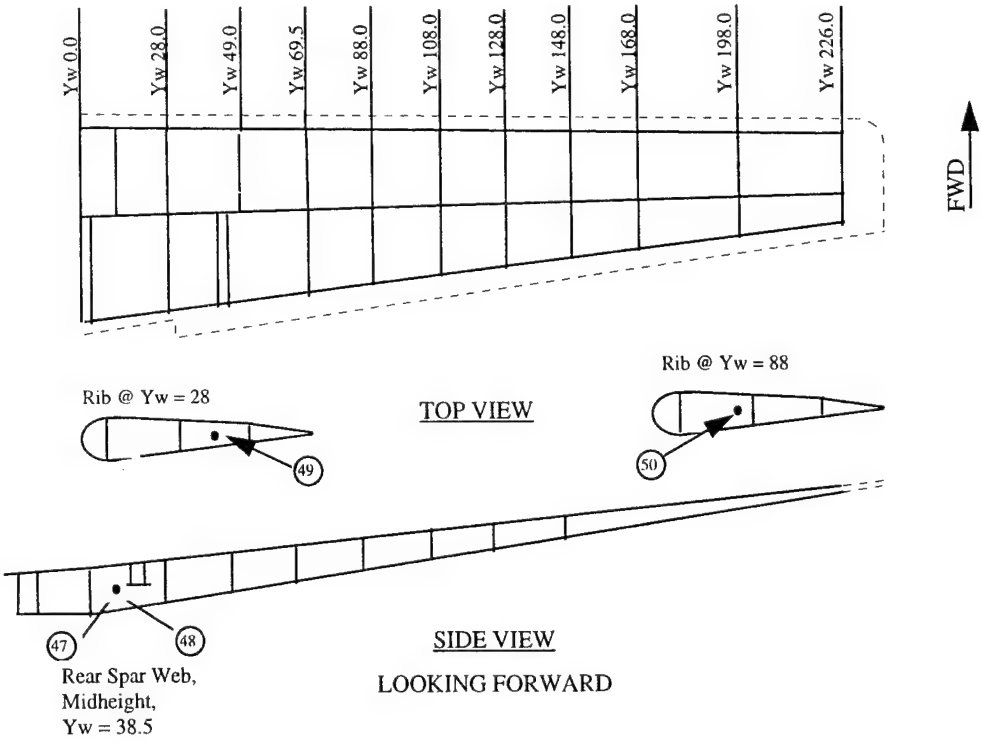


FIGURE 4-13. SUBSTRUCTURE ANALYSIS LOCATIONS

Five high-strain points were selected for the upper skin, as shown in table 4-5; 8 high-strain points were selected for the lower skin, as shown in table 4-6; and 4 were selected for the substructure, as shown in table 4-7. In order to represent the total wing box structure, other lower-strain portions of the wing box were defined. Figure 4-14 shows the total wing box structure consisting of upper skin, lower skin, and substructure (spars and ribs) at indicated locations. The total upper skin area was represented by 4 high-strain panels and 16 low-strain panels. In a similar manner, the lower skin was represented by 6 high-strain panels and 14 low-strain panels, while the substructure contained 4 high-strain and 49 low-strain areas. In figure 4-14, U indicates upper skin, L indicates lower skin, and S represents substructure. The numbering indicates the point identification number used in the data input file, (appendix C). Failure probabilities associated with these 17 high-strain locations and 79 low-strain locations thus represent the total probability of structural failure of the wing box. This makes the total number of locations on the Lear Fan wing box analyzed equal to 96.

The low-strain regions are accounted for by assuming an applied strain of 1,900 μ in/in for both the upper and lower skin and 1,500 μ in/in for the substructure with an associated static test ultimate allowable strain of 5,000 μ in/in. These values were representative of the highest recorded strains, under 2,000 μ in/in, in the upper and lower skin and substructure.

TABLE 4-5. HIGH-STRAIN ANALYSIS POINTS—UPPER SKIN

Condition ~ Symmetric Up Bending

Point ID	Static Test Max. μ in/in(1)	Allowable μ in/in	M.S. at Static Test	Design Limit Strain(2)	M. S. at Static Test RTD(3)
1	-3,390	-5,000	0.48	-1,982	+0.68
2	-2,200	-5,000	1.27	-1,287	+1.59
3	-2,820	-5,000	0.77	-1,649	+1.02
4	-2,950	-5,000	0.70	-1,725	+0.93
5	-2,000	-5,000	1.50	-1,170	+1.85

Notes: (1) Strain measurements taken at 94% of Ultimate Load

(2) Limit strain = $\frac{[\text{Static Test Max. Strain}]}{(1.5)(1.14)}$

(3) For restricted aircraft (94% of Ultimate Load)

TABLE 4-6. HIGH-STRAIN ANALYSIS POINTS—LOWER SKIN

Condition ~ Symmetric Up Bending

Point ID	Static Test Max. μ in/in(1)	Allowable μ in/in	M.S. at Static Test	Design Limit Strain (2)	M. S. at Static Test RTD(3)
22	+2,700	5,000	+0.85	+1,579	+1.11
23	+2,270	5,000	+1.20	+1,327	+1.51
24	+2,420	5,000	+1.07	+1,415	+1.36
25	+2,413	5,000	+1.07	+1,411	+1.36
26	+2,370	5,000	+1.11	+1,387	+1.41
27	+2,000	5,000	+1.50	+1,170	+1.85
28	+2,770	5,000	+0.81	+1,620	+1.06
29	+2,260	5,000	+1.21	+1,322	+1.52

Notes: (1) Strain measurements taken at 94% of Ultimate Load

(2) Limit strain = $\frac{[\text{Static Test Max. Strain}]}{(1.5)(1.14)}$

(3) For restricted aircraft (94% of Ultimate Load)

TABLE 4-7. HIGH-STRAIN ANALYSIS POINTS—SUBSTRUCTURE

Condition ~ Symmetric Up Bending

Point ID	Static Test Max. μ in/in[1]	Allowable μ in/in	M.S. at Static Test	Design Limit Strain[2]	M. S. at Static Test RTD[3]
44	-3,010	-5,000	+0.66	-1,760	+0.89
45	+2,120	+5,000	+1.36	+1,240	+1.69
46	-2,050	-5,000	+1.44	-1,199	+1.78
47	-2,040	-5,000	+1.45	-1,193	+1.79

Notes: (1) Strain measurements taken at 94% of Ultimate Load

(2) Limit strain $= \frac{[\text{Static Test Max. Strain}]}{(1.5)(1.14)}$

(3) For restricted aircraft (94% of Ultimate Load)

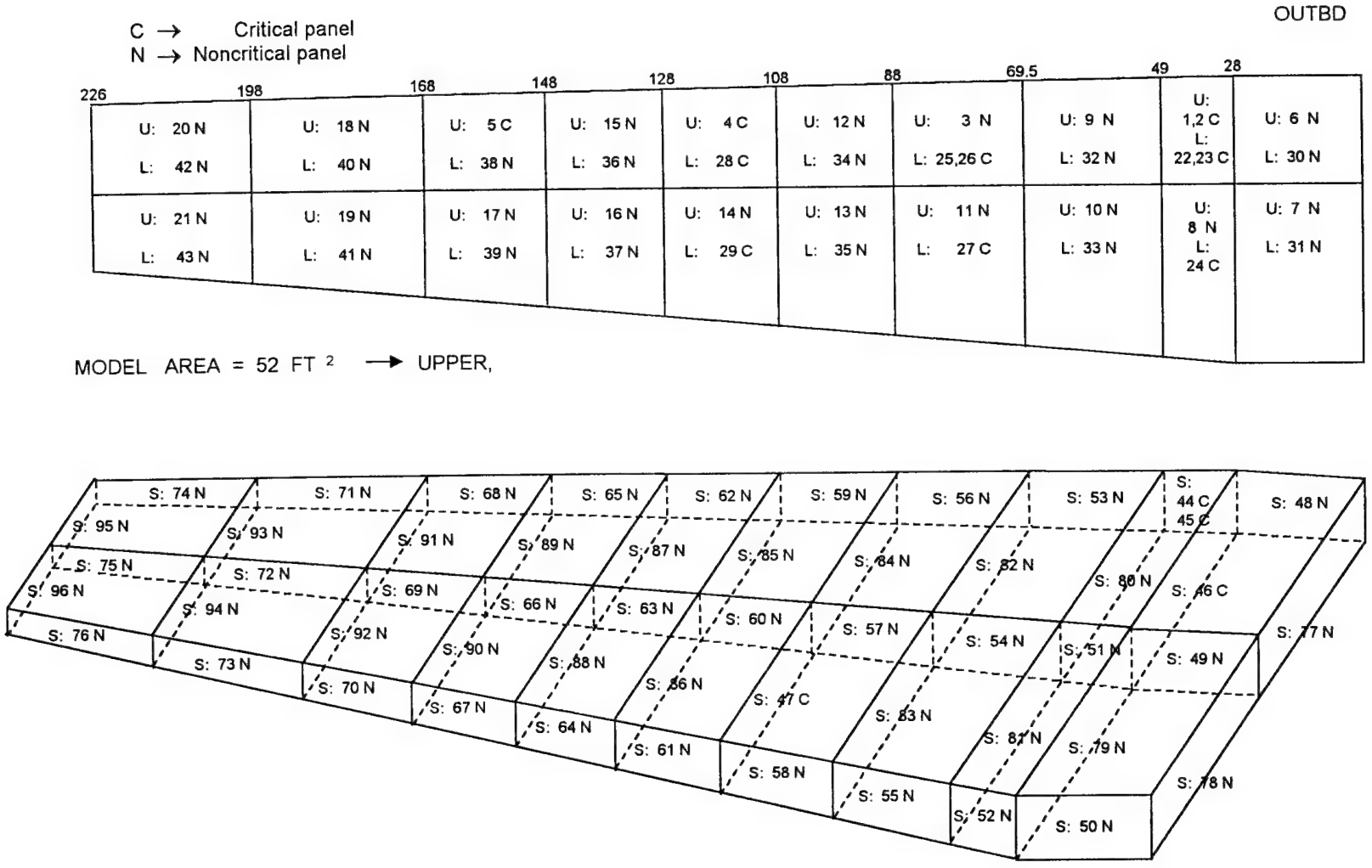


FIGURE 4-14. RISK ASSESSMENT LOCATION INFORMATION

4.4 FLIGHT TEMPERATURES.

The altitudes and times of the individual segments of each of the mission profiles were used to obtain the time distribution of structural temperature in flight. Table 4-8 shows the profile of the individual segments of the three missions. A conservative assumption was made that the structure has been heated on the ground and does not reach equilibrium with ambient until after the first few minutes (or equivalently, the first two segments) of each mission. Segment 1 was assumed to be 160°F and segment 2 assumed to be 100°F. After this point, the structural temperature was assumed to equal the standard day ambient temperature at altitude. When the segment duration (in minutes) is multiplied by the number of flights (see discussion on p. 4-9), the total lifetime exposure time (in minutes) results and these are given in the last column.

TABLE 4-8. MISSION SEGMENT TIME AND TEMPERATURE

Mission Number	Segment Number	Type	Mean Altitude (Ft)	Time (Min)	Ambient Temp Std.Atmospher e (°R)	Struct Temp °F (=°R-459.4)	Total Lifetime Minutes
1	1	Climb	2500	1.40	509.8	160.0	3,522.6
1	2	Climb	7500	1.50	491.9	100.0	3,774.2
1	3	Climb	12500	1.60	474.1	14.7	4,025.8
1	4	Climb	17500	1.70	456.2	-3.2	4,277.4
1	5	Cruise	20000	9.50	447.4	-12.0	23,903.2
1	6	Cruise	20000	9.50	447.4	-12.0	23,903.2
1	7	Descent	6000	6.00	497.3	37.9	15,096.8
1	8	Approach	1000	6.00	515.1	55.7	15,096.8
2	1	Climb	2500	1.15	509.8	160.0	10,052.8
2	2	Climb	7500	1.30	491.9	100.0	11,364.1
2	3	Climb	12500	1.45	474.1	14.7	12,675.3
2	4	Climb	17500	1.90	456.2	-3.2	16,609.0
2	5	Climb	22500	2.35	438.5	-21.0	20,542.7
2	6	Climb	27500	3.30	420.1	-39.3	28,847.3
2	7	Cruise	30000	21.15	411.7	-47.7	184,884.7
2	8	Cruise	30000	21.15	411.7	-47.7	184,884.7
2	9	Descent	16000	6.20	461.6	2.2	54,197.9
2	10	Cruise	10000	6.00	483.0	23.6	52,449.6
2	11	Descent	6000	3.00	497.3	37.9	26,224.8
2	12	Approach	1000	6.00	515.1	55.7	52,449.6
3	1	Climb	2500	1.15	515.1	160.0	1,164.7
3	2	Climb	7500	1.30	509.8	100.0	1,316.6
3	3	Climb	12500	1.45	474.1	14.7	1,468.5
3	4	Climb	17500	1.90	456.3	-3.2	1,924.2
3	5	Climb	22500	2.35	438.5	-21.0	2,380.0
3	6	Climb	27500	3.30	420.1	-39.3	3,342.0
3	7	Climb	32500	4.60	402.8	-56.6	4,658.6
3	8	Cruise	35000	55.00	393.9	-65.5	55,700.7
3	9	Cruise	35000	55.00	393.9	-65.5	55,700.7
3	10	Descent	22500	8.25	438.4	-21.0	8,355.1
3	11	Cruise	10000	6.00	483.0	23.6	6,076.4
3	12	Descent	6000	3.00	497.3	37.9	3,038.2
3	13	Approach	1000	6.000	515.1	55.7	6,076.4
				Total Lifetime Flight Minutes			899,984.5
				Total Lifetime Flight Hours			15,000

The flight temperature lifetime distribution is obtained from the previous data and is summarized in table 4-9 as mean temperature in a given time interval versus time. By creating a discrete probability distribution from this data, the temperature is modeled independently of the time in the mission. That is, temperature is randomly chosen within each Monte Carlo trial; these trials do not conform to any time sequence of any particular mission. Rather, the Monte Carlo randomly draws a temperature with probability equal to the ratio of total lifetime minutes at that temperature to total lifetime minutes. This is a conservative approach in that the methodology allows for the situation of high-g maneuvers coupled with high temperature, which is highly unusual for this aircraft.

TABLE 4-9. FLIGHT LIFETIME TEMPERATURE DISTRIBUTION

Mean Structure Temperature(°F) in Interval	Total Lifetime Minutes	Total Lifetime (%)
-65.5	111,401.4	12.4
-56.6	4,658.6	0.5
-47.7	369,769.5	41.1
-21.0	52,732.1	5.9
-12.0	47,806.5	5.3
-3.2	31,621.5	3.5
2.2	54,197.9	6.0
14.7	18,625.3	2.1
23.6	58,526.0	6.5
32.5	1,468.5	0.2
37.9	44,359.8	4.9
55.7	73,622.8	8.2
100.0	16,454.8	1.8
160.0	14,740.1	1.6
Total	899,984.5	100.0

4.5 LEAR FAN WING BOX ANALYSIS/RESULTS.

The 17 high-strain points of the Lear Fan wing box presented in tables 4-5 through 4-7, along with the other 79 low-strain points described in section 3.3, were used with statistical modeling of material strength (per failure mode), manufacturing defects, operational defects, operations-induced damage, moisture absorption, and gust loading to obtain the NGCAD probabilistic design input file presented in appendix C. These data, as incorporated into the probabilistic design methodology of figure 3-3, resulted in a predicted structural failure risk of the wing box.

The analysis was performed using actual static test strain measurements. Critical upper and lower skin locations were defined to be in the vicinity of fastener holes. From an engineering experience, for a wing skin/spar cap joint subjected to spanwise loading, typical fastener bearing loads are expected to be in the neighborhood of 25 percent of the total net section loads, which corresponds to an effective net section stress concentration factor of approximately 2.16. This

value was based on the use of the Bolted Joint Stress Field Model (BJSFM) computer program developed by Garbo et al [17]. The operating stress was therefore scaled by a factor of 2.16 to account for bearing/bypass stress concentrations in the probabilistic model.

A key step in the probabilistic analysis is to define what is meant by system failure. For this analysis, it was assumed that there was no load path redundancy or load redistribution and all locations and failure modes were critical to system survival. Therefore system failure was defined as maximum stress exceeding material strength at any location for any failure mode.

Two separate cases were analyzed for the upper skin of the restricted aircraft; failure was defined as an incident in which the strain at any location on the upper skin exceeded either: (1) the critical buckling strain or (2) the critical compression strain. In the case of buckling, failure was restricted to local buckling and postbuckling was not considered. In the case of the lower skin and substructure, these were modeled only as strength-critical, the lower skin failing in tension (denoted in tables 4-10 through 4-12 as TS) and the substructure in compression (denoted in the tables as CS) except at location 45, which was modeled using the TS failure mode.

Tables 4-10 through 4-12 show individual component [upper skin (US), lower skin (LS), substructure (SS), high-strain (H), and low-strain (L)] risks and are summed to indicate the total risk for one wing. Table 4-10 shows the total wing structural failure risk using the buckling mode (denoted CM) on the upper skin. The total wing probability of failure was 1.14×10^{-6} and was driven by the upper skin risk. Tables 4-11 and 4-12 show wing structural failure risk for the restricted and unrestricted aircraft, respectively, when the upper skin is modeled as being compression strength critical. The predicted total wing single flight probability of failure for the restricted aircraft was 1.17×10^{-9} and for the unrestricted aircraft was 1.98×10^{-9}. Because the upper skin contained the highest strains, coupled with the lower residual strength after damage (compared to tension failure mode), the total wing probability of failure was driven by the upper skin.

TABLE 4-10. SINGLE-FLIGHT PROBABILITIES OF FAILURE: RESTRICTED AIRCRAFT, UPPER SKIN BUCKLING CRITICAL

Location	Failure Modes			Single Flight Probability of Failure
	CM	CS	TS	
Total	1.1419E-6	0.229E-10	2.111E-10	1.1420E-6
USH	1.1409E-6			1.1409E-6
USL	0.0010E-6			0.0010E-6
LSH			1.775E-10	1.775E-10
LSL			0.336E-10	0.336E-10
SSH		0.056E-10	≅0	0.056E-10
SSL		0.173E-10		0.173E-10

Single-Flight Probabilities of Failure, Restricted Aircraft at 3.29 g's
Number of Flights = 12275 Number of Years = 33.63

TABLE 4-11. SINGLE-FLIGHT PROBABILITIES OF FAILURE: RESTRICTED AIRCRAFT, UPPER SKIN COMPRESSION CRITICAL

Location	Failure Modes		Single Flight Probability of Failure
	CS	TS	
Total	9.609E-10	2.111E-10	1.172E-9
USH	9.368E-10		9.368E-10
USL	0.012E-10		0.012E-10
LSH		1.775E-10	1.775E-10
LSL		0.336E-10	0.336E-10
SSH	0.056E-10	≅0	0.056E-10
SSL	0.173E-10		0.173E-10

Single-Flight Probabilities of Failure, Restricted Aircraft at 3.29 g's
Number of Flights = 12275 Number of Years = 33.63

TABLE 4-12. SINGLE-FLIGHT PROBABILITIES OF FAILURE: UNRESTRICTED AIRCRAFT, UPPER SKIN COMPRESSION CRITICAL

Location	Failure Modes		Single-Flight Probability of Failure
	CS	TS	
Total	1.731E-9	0.253E-9	1.984E-9
USH	1.701E-9		1.701E-9
USL	0.002E-9		0.002E-9
LSH		0.216E-9	0.216E-9
LSL		0.037E-9	0.037E-9
SSH	0.008E-9	≅0	0.008E-9
SSL	0.020E-9		0.020E-9

Single-Flight Probabilities of Failure, Unrestricted Aircraft at 3.50 g's
Number of Flights = 12275 Number of Years = 33.63

The primary purpose of this analysis was to illustrate the capability of the analysis for quantifying structural risk in terms of a probability of failure. The analysis methodology uses the Monte Carlo simulation to adjust both the stress and strength distributions prior to numerical integration. The accuracy of the answer depends on the quality of the input data and the mechanics of the analysis.

This Lear Fan analysis is as complete and accurate as possible given the constraints under which it was conducted. It should not be considered an absolute result due to the following data considerations.

The input data is the focus of most of the concerns regarding output accuracy. This begins with the use of measured stress instead of predicted stress verified by measurement. The use of measured stresses restricts analysis locations to specific points. Definition of failure was also hypothesized on the basis of measurements and responses made during the Lear static test. This yields a very accurate representation of the structure at the failure location but does not provide all that is needed for the structure remote from this location. Conservative assumptions were used to characterize these remote areas.

The input data used to characterize basic material mechanical properties was derived from coupon tests. Statistically significant data was available only for the dry condition. As stated previously, moisture effects for compression strength were represented by using residual strength factors developed by NGCAD for a similar material system.

Manufacturing defect data was unavailable from the original manufacturer. These data were supplied from NGCAD experience. There are no industry standards for characterizing defect parameters; defect occurrence data are usually available (discrepancy reports such as withholding tags) but not in a form directly applicable as probabilistic input. Additional effort was required to gather and analyze the data for proper characterization. This included frequency of occurrence, type of defect, and effect on structural strength.

A similar discussion can be applied to operational defects. This data was unavailable as well; the characterization of operational damage rates and severity was done by using information obtained from visits to airline maintenance facilities and naval aviation depots.

To put this probabilistic methodology into practice obviously requires careful planning, preparation, measurement, and confidence building. It is envisioned that its ultimate use or application would be more accepted through a phased introduction approach. Initially, it is believed that it can be most helpful as a design check or information tool. This could progress into more strategic value where it would be used as a guide for design and certification criteria. Ultimately the methodology will be used as a primary tool for certification of composite airframe components. Considerable development, refinement, and verification must occur before this final phase is appropriate.

5. DATABASE DEVELOPMENT.

The foundation of the probabilistic methodology is the compilation of the design parameters and their distributions. Therefore, the characterization of the operational damage, location, and relevant flight hours was done by analyzing the data obtained during visits to airline maintenance facilities and Naval aviation depots. Data obtained from visits to American Airlines, Delta Airlines, United Airlines, the North Island Naval Aviation Depot (NADEP), and from communications with De Havilland Aircraft, Inc. provided much information which is summarized in this section. Since hail storms pose a serious threat to composites, the database survey included data on the frequency of exposure to hail. Geometric structural details are incorporated including panel thickness, material, and edge support.

It was additionally intended to collect information on the relationship between observed and measured damage and applied impact energy. However, none of the facilities had any data on this subject.

Appendix D contains the International Air Transport Association (IATA) questionnaire on composite structure maintenance which represents the response of 19 worldwide operators. These small, medium, and large size operators maintain a total of 2100 commuter, short-, medium-, and long-range commercial aircraft of which 1000 aircraft have advanced composite structure.

TABLE 5-1. IATA GENERAL INFORMATION SUMMARY

Number of operators	19
Number of aircraft	2100
Number of aircraft containing composite structure	1000
Repair Facilities	
Operators having autoclave facilities	4
Operators having 2 to 7 m ovens	10
Operators having hot bonding units	15
Repair Instructions	
Use Structural Repair Manual and operator specific procedures	9
Use Structural Repair Manual only	6
Use operator specific procedures only	4
Composite Repairs	
Percent glassfiber reinforced composite structure	90%
Percent advanced composite structure	10%
Wet lay-up is used as a permanent repair method	80%
Repair Deferment Average Time	
Large damage (safety of flight not compromised)	1 Flight
Small cracks on secondary structure	several months

Table 5-2 presents a summary of information obtained from two airlines and DeHavilland Aircraft, Inc.; table 5-3 presents occurrence rates per million flight hours.

TABLE 5-2. SUMMARY OF AIRLINE INFORMATION

	American Airlines	De Havilland Aircraft, Inc.	United Airlines	Total
Flight hours	2,005,896	117,134	1,691,775	3,814,805
Hail storms	5	5	1	11
Lightning strikes	60	8	51	119
Bird strikes	0	4	3	7
Maintenance induced damages	585	312	491	1484

TABLE 5-3. EVENT OCCURRENCE RATES

	Occurrences Per 1,000,000 Flight Hours
Hail storm	3
Lightning strike	31
Bird strike	2

Damage report information is summarized in tables 5-4 and 5-5; damage type is presented per million flight hours in table 5-6.

TABLE 5-4. DAMAGE CAUSE FREQUENCY AND OCCURRENCE RATES

Damage Cause Frequency	Frequency	Occurrences Per 10^6 Flight Hours
Lightning	7%	76
Bird strike and hail	8%	86
Moisture and chemical	30%	324
Runway stones	8%	86
Maintenance induced damage	36%	389
Other	11%	119
Total	100%	1081

TABLE 5-5. DAMAGE TYPE FREQUENCY

Damage Type	Frequency	Damage Size (inches)		
		<1.5	1.5 to 3.0	>3.0
Hole damage	35%	18%	12%	5%
Delamination	45%	5%	14%	27%
Cracks	10%	3%	3%	4%
Subtotal	90%	26%	29%	36%
Other	10%	N/A	N/A	N/A
Total	100%			

TABLE 5-6. DAMAGE TYPE OCCURRENCE RATES

	Occurrences per 1,000,000 Flight Hours Damage Size (inches)			
Damage Type	<1.5	1.5 to 3.0	>3.0	Total
Holes	189	132	57	378
Delamination	49	146	292	487
Cracks	32	32	43	107
Total	270	310	392	972

6. CONCLUSIONS.

The methodology evaluation activity carried out under this effort indicated that several approaches are available for probabilistic risk assessment of composite aircraft design, from the limited European effort which focused on probabilistic maintenance scheduling for Aerospatiale and Airbus aircraft to the more broadly based methodology developed by Northrop Grumman Commercial Aircraft Division (NGCAD) as well as that of NASA Lewis developed under Chamis.

By carrying out the evaluation of the Lear Fan composite wing, this effort successfully demonstrated the suitability of the NGCAD approach for assessing the level of risk inherent in a typical composite primary structural aircraft component. Within the limitations of the effort, especially in regard to limitations of available input data, results of the evaluation indicated that the risk of failure is on the order of 1E-09 per flight which corresponds to a very small probability of failure (1E-05) for a typical aircraft during its design life.

Limitations of the current analysis of the Lear Fan were

- dependence of the present analysis on measured strains obtained from component test data rather than strains obtained from structural analysis, which prevented risk assessment in regions other than those where the measured strains were available.

- lack of manufacturing defect and environmental effects data on the Lear Fan, leading to the need for assuming representative data for these effects from other aircraft.

- lack of aircraft storage temperature and humidity profiles for assessment of laminate moisture absorption, resulting in assuming a worst-case environment corresponding to Guam.

In addition to the limitations just described, the database on typical operationally induced defects in composite aircraft components (effects of dropped tools, runway debris, etc.) is not adequate at the present time. Although the database development effort of section 5 did provide a considerable amount of useful data concerning the frequency and general type of operationally induced damage, the associated damage severity was not quantified. Continuing effort over the next several years will be needed to accumulate adequate data on defects caused by service operations on composite aircraft.

On the basis of the NGCAD and other approaches investigated in this effort, the probabilistic design process, with the noted gaps and limitations allowed for, is sufficiently well defined for available computational methods and data requirements to make the probabilistic approach to design of composite airframe structures feasible.

7. RECOMMENDATIONS FOR FOLLOW-ON WORK.

While this reporting phase of the program established a solid background for probabilistic evaluations by the FAA, for probabilistic technology to become more widely understood, Northrop Grumman Commercial Aircraft Division (NGCAD) recommends the following additional tasks be considered.

- Evaluation of Comparable Methodologies. The results obtained from Phase I in 1994 disclosed that the Probabilistic Design approach put forth by Christopher Chamis at NASA-Lewis Research Center overlaps similar work by NGCAD. The NGCAD approach was utilized in Phase I to assess the reliability of the Lear Fan wing box, and the Chamis method is recommended to be applied to the same structure to obtain another solution for comparison. It is recommended that other efforts, such as those by Aerospatiale, be further evaluated in order to discuss critical aspects of these technologies.

- Aircraft Structure Risk Drivers. In order to aid the FAA in identifying critical risk drivers, it is recommended that the Lear Fan probabilistic analysis performed in Phase I be extended to identify the magnitude of the contribution of various criteria to the probability of failure.

  To assess the goal for target reliability and fleet historical values obtained from records kept by the airlines, FAA Accident Investigation Board and National Transportation Safety Board (NTSB) would be evaluated to indicate what probability of failure experience has been to date.

- Develop a PC version of NGCAD's Computer Analysis Program. NGCAD recommends the development and delivery to the FAA of a version of the Probabilistic Methodology software configured to run on a dedicated PC within a Microsoft Windows NT operating system including procedures, executable files, and a user's manual.

- Certification. It is recommended that this task address probabilistic certification and scale-up effects.

- Methodology Enhancements. Develop a means to account for correlation between maneuver loads and structure temperature. Develop the capability to model structure temperature representing random variations of ambient distributions.

- Comparison of NGCAD and TsAGI Approaches to Probabilistic Design of Composite Structures. Comparison of the NGCAD approach with the approach being developed by the Russian Aerohydrodynamic Institute (TsAGI) under FAA support is of interest. Use of the two approaches on the same case for comparison of computational efficiency and quality of results from the two methods is desirable.

8. REFERENCES.

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APPENDIX A—VOUGHT COMMERCIAL QUESTIONNAIRE

VOUGHT AIRCRAFT COMPANY, DALLAS, TX

STRUCTURAL FAILURES DATA COLLECTION TASK

UNITED AIRLINE VISIT - SAN FRANCISCO

29 JULY 1994

1. What are the intervals for letter checks?
2. How many airplanes are in your fleet?
3. How many flights per month?
4. How many total flight hours per month?
5. After each check, how many composite repairs are needed per A/C?
6. What percent repairs are due to induced/accidental/FOD/etc., damage?
7. What portion of repairs due to edge delamination of doors and panels?
8. What percent of induced damage due to: (classify type of damage and typical component accidental impact with tools, vehicles)

    Debris blown by engines or kicked up by tires?

    Wear from moving parts?

    Fluids?

    Other induced damage?
9. How often do you get bird strike damage?
10. How often do you get hail damage?
11. How often do you get lightning damage?
12. How do you repair graphite epoxy?
13. How do you repair graphite Kevlar?
14. How do you repair epoxy Kevlar?
15. How do you repair epoxy honeycomb?
16. What rules do you apply for deferred maintenance?

APPENDIX B—VOUGHT NADEP QUESTIONNAIRE

FAA PROBABILISTIC DESIGN STUDY—COMPOSITE PARTS

NAVAL AVIATION DEPOT (NADEP):

AIRCRAFT: _____

FLEET INFORMATION

1. HOW MANY AIRPLANES WERE IN THE FLEET IN 1993? _____

   TOTAL 1993 OPER HOURS _____

2. About how many flights per airplane per month? _____

3. About how many hours per flight per aircraft? _____

OPERATING ENVIRONMENT

4. Afloat _____ Ashore _____ Hangared _____

AIRCRAFT

5. Please identify the composite parts on the _____ aircraft on the attached cutaway, or by some other method.

6. What composite materials are used? (graphite/epoxy, Kevlar, etc.)

   _____

7. What types of construction are used? (honeycomb, solid laminate, metal bonded, etc.)

   _____

FAILURES OF COMPOSITE PARTS

8. About what percentage of the repairs were necessary because of inherent failures? _____ induced failures? _____

9. What were the major inherent failure modes and their percentages?

   _____

   _____

   _____

10. What were the major induced failure modes and their percentages?

_____

_____

_____

11. How often has the fleet been damaged by

bird strike _____ hail _____ lightning _____

INSPECTION

12. How are composite parts inspected?

x-ray eddy current ultrasonic tap test other

REPAIR

13. About how many composite repairs are performed each month? _____

14. About what percentage of all composite repairs for the fleet are performed at NADEP? _____

15. Which composite materials and types of construction are the hardest to repair?

_____

easiest to repair? _____

16. What percentage of total maintenance events are:

Simple _____ Medium _____ Difficult _____

17. How many hours does it take to fix repairs:

Simple _____ Medium _____ Difficult _____

```
MONTE INPUT DATA FILE
TITLE: LFA -  Restricted A/C; Upper CS, Upper/Lower RT, 150% DLL, All Effects On
DATE : 11 JUNE 1996
.
01        Main counters                                              PARAMS.INC
      96  Number of locations used                    (Max of 500)
       1  Number of thicknesses used                   (Max of 10)
   1   1  Beginning and ending thickness sensitivity
       2  Number of failure modes used                 (Max of 6)
   20000  Number of Monte Carlo loop iterations  (Max of 20,000)
.
02        Flight and Lifetime Information                            GENERIC.INC
 33.630024 Analysis: Start time, in years
 33.630024 Analysis: End   time, in years
  446.03   Average number of flight hours per year
   1.222   Average number of hours per flight
.
03        Output Generation Control Flags                            GENERIC.INC
.
>>>>>>>>> Output details for each PF computed
N         Flag: "Y" or "N"
.
>>>>>>>>> Output details for each AVEPF computed
N         Flag: "Y" or "N"
.
>>>>>>>>> "Save graphics data" indicator flags
       0  Count of specification lines
.
>>>>>>>>> Location Sets for Subtotal Output
       6  Count of sets (Max of 20)
USH       Location Set #1: Upper Skin High Strain Locations
       1  Count of range specification lines for location set #1
   1   5
USL       Location Set #2: Upper Skin Low Strain Locations
       1  Count of range specification lines for location set #2
   6  21
LSH       Location Set #3: Lower Skin High Strain Locations
       1  Count of range specification lines for location set #3
  22  29
LSL       Location Set #4: Lower Skin Low Strain Locations
       1  Count of range specification lines for location set #4
  30  43
SSH       Location Set #5: Substructure High Strain Locations
       1  Count of range specification lines for location set #5
  44  47
SSL       Location set #6: Substructure Low Strain Locations
       1  Count of range specification lines for location set #6
  48  96
.
04        Miscellaneous Information                                  GENERIC.INC
NEW       Program Version ("OLD" or "NEW")
 1324153  Random number generator seed (should be a fairly large number)
YES       The DLL truncation usage indicator flag ("YES" or "NO")
150.000   DLL Truncation percent
  1.000   The averaging class value for computing AVEPFs
    1.5   The design factor (usually 1.5)
.
>>>>>>>>> Major Aircraft Component Definitions
```

```
N         Print/Pause flag for effective densities & execution control option
N         Write NAC effective density table to MONTE output file
3         Count of NACs
UPPER     Lear Fan Upper Skin
100.0     Weight of Upper Skin
1         Count of location inclusion specification lines
1   21    Location range 1
LOWER     Lear Fan Lower Skin
100.0     Weight of Lower Skin
1         Count of location inclusion specification lines
22  43    Location range 1
SUBST     Lear Fan Substructure
100.0     Weight of Substructure
1         Count of location inclusion specification lines
44  96    Location range 1
.
=========+=========+=========+=========+=========+=========+=========+=========+
05        Location Information                                          FMLOC.INC
---------+---------+---------+---------+---------+---------+---------+---------+
.
>>>>>>>>> Baseline thicknesses for each location  (thk  j1 j2)
0.200     001  021
0.100     022  096
.
>>>>>>>>> Area of each location in square feet  (area   j1  j2)
1.35      001  002
2.20      003  003
2.15      004  004
1.88      005  005
3.69      006  006
5.05      007  007
3.50      008  008
2.50      009  009
3.13      010  010
2.64      011  011
2.30      012  012
2.64      013  013
2.36      014  014
2.01      015  015
2.08      016  016
1.88      017  017
2.60      018  018
2.50      019  019
2.24      020  020
1.94      021  021
1.35      022  023
3.50      024  024
1.10      025  026
2.64      027  027
2.15      028  028
2.36      029  029
3.69      030  030
5.05      031  031
2.50      032  032
3.13      033  033
2.30      034  034
2.64      035  035
2.01      036  036
2.08      037  037
1.88      038  039
2.60      040  040
2.50      041  041
2.24      042  042
1.94      043  043
0.40      044  045
0.80      046  046
```

```
0.60     047  047
1.20     048  050
0.80     051  052
0.70     053  058
0.60     059  061
0.50     062  067
0.40     068  073
0.30     074  076
0.80     077  077
1.00     078  079
0.80     080  080
0.90     081  081
0.60     082  082
0.70     083  083
0.50     084  084
0.40     085  085
0.50     086  086
0.30     087  087
0.40     088  088
0.30     089  090
0.20     091  091
0.10     095  096
.
>>>>>>>>> Number of 45 degree plies per location  (# 45 Deg plies    j1  j2)
 7       001  021
14       022  096
.
--------+--------+--------+--------+--------+--------+--------+--------+
06       Thickness Sensitivity Information                      FMLOC.INC
---------+--------+--------+--------+--------+--------+--------+--------+
.
>>>>>>>>> Thickness sensitivity adjustment percentages (i.e. 105 100 90 85 ...)
100
.
--------+--------+--------+--------+--------+--------+--------+--------+
07       Failure Mode Information                               FMLOC.INC
---------+--------+--------+--------+--------+--------+--------+--------+
.
>>>>>>>>> Count of failure mode usage definition blocks
2
.
>>>>>>>>> Failure Mode Usage Definition Block 1
Y N          Failure Mode usage pattern
3            Count of range specificaton lines
001  021     Beginning and ending locations for range
044  044         "    "   "       "     "   "
046  096         "    "   "       "     "   "
.
>>>>>>>>> Failure Mode Usage Definition Block 2
N Y          Failure Mode usage pattern
2            Count of range specificaton lines
022  043     Beginning and ending locations for range
045  045         "    "   "       "     "   "
.
>>>>>>>>> Failure mode type codes (one per line)
CS
TS
.
--------+--------+--------+--------+--------+--------+--------+--------+
08       Margins of Safety                                      FMLOC.INC
---------+--------+--------+--------+--------+--------+--------+--------+
.
>>>>>>>>> Baseline Margins of Safety, at Room Temp (value    j1 j2  k1 k2)
  0.69     001  001  1  1
  1.59     002  002  1  1
  1.02     003  003  1  1
```

```
 0.93     004  004  1  1
 1.85     005  005  1  1
 2.00     006  021  1  1
 1.11     022  022  2  2
 1.81     023  023  2  2
 1.36     024  024  2  2
 1.36     025  025  2  2
 1.41     026  026  2  2
 1.85     027  027  2  2
 1.06     028  028  2  2
 1.82     029  029  2  2
 2.00     030  043  2  2
 0.89     044  044  1  1
 1.69     045  045  2  2
 1.78     046  046  1  1
 1.79     047  047  1  1
 2.80     048  096  1  1

>>>>>>>>> Alternate Margins of Safety (Must include this line)

09       Material Strength Allowables                          FMLOC.INC

>>>>>>>>> Material Strength Allowables (Value, BegLoc, EndLoc, BegFM, EndFM)
 5000.0    001  021    1  1
 5000.0    022  043    2  2
 5000.0    044  044    1  1
 5000.0    045  045    2  2
 5000.0    046  096    1  1

>>>>>>>>> Material Strength Knock-Down Factors (Val, BLoc, ELoc, BFM, EFM)
 1.000     001  096    1  2

10       Maximum Operating STRESS Distribution Information     DSTRIB.INC

>>>>>>>>> Count of distribution definition blocks
2

>>>>>>>>> Distribution definition block 1 - Upper and Lower Skin RT=2.16 effect
LOGNORMAL
-0.29634     0.53677   2.16000
    3.29             The number of Gs that are equal to 100% DLS
1.222                Hours associated with the distribution
2                    Count of range specification lines
  1   21     1  1    [Beg Loc, End Loc,  Beg FM, End FM]
 22   43     2  2    [Beg Loc, End Loc,  Beg FM, End FM]

>>>>>>>>> Distribution definition block 2
LOGNORMAL
-1.06645     0.53677   1.0000
       3.29          The number of Gs equal to 100% DLS
   1.222             Flight hours associated with the distribution
3                    Count of range specification lines
 44   44     1  1    [Beg Loc, End Loc, Beg FM, End FM]
 45   45     2  2    [Beg Loc, End Loc, Beg FM, End FM]
 46   96     1  1    [Beg Loc, End Loc, Beg FM, End FM]

11       Material STRENGTH Distribution Information (NORMAL)   DSTRIB.INC

>>>>>>>>> Count of distribution definition blocks
4
```

```
>>>>>>>>> Distribution definition block: Upper Skin: Compression Failure (Tape)
NORMAL
11560.0      1368.0
1                    Count of range specification lines
  1   21     1  1    [Beg Loc, End Loc,  Beg FM, End FM]
.
>>>>>>>>> Distribution definition block: Lower Skin: Tension Failure (Tape)
NORMAL
10550.0     1336.0
1                    Count of range specification lines
 22   43     2  2    [Beg Loc, End Loc,  Beg FM, End FM]
.
>>>>>>>>> Distribution definition block: Substructure: Comp. Failure (Fabric)
NORMAL
9940.0       1315.0
2                    Count of range specification lines
 44   44     1  1    [Beg Loc, End Loc,  Beg FM, End FM]
 46   96     1  1
.
>>>>>>>>> Distribution definition block: Substructure: Tension Failure (Fabric)
NORMAL
6900.0       836.0
1                    Count of range specification lines
 45   45     2  2    [Beg Loc, End Loc,  Beg FM, End FM]
.
-------+---------+--------+---------+---------+---------+---------+---------+
14       Gust Effects Information                                  GUST.INC
---------+---------+--------+---------+---------+---------+---------+---------+
Y        Usage Indicator flag (Y or N)
0.1      The Probability of Gust Occurring     (via [0,1] uniform)
0.0      Probability that the gust is downward (via [0,1] uniform)
.
>>>>>>>>> Downward Gust Distribution (produces a negative shift)
LOGNORMAL  Distribution Type
-6.2688  1.9018  0.1458
.
>>>>>>>>> Upward Gust Distribution (produces a positive shift)
UNIFORM    Distribution Type
0.10     0.30
.
---------+---------+--------+---------+---------+---------+---------+---------+
15       Manufacturing Defects Information                         MFGDEF.INC
---------+---------+--------+---------+---------+---------+---------+---------+
Y        Usage Indicator flag (Y or N)
.
>>>>>>>>> Count and types of Manufacturing defects
5        The number of manufacturing type being used (Max of 5)
HOLE_DEF  Manufacturing defect 1 type name (Max of 8 chars)
LAMINATE  Manufacturing defect 2 type name (Max of 8 chars)
IMPACT    Manufacturing defect 3 type name (Max of 8 chars)
WAVINESS  Manufacturing defect 4 type name (Max of 8 chars)
TOLERNCE  Manufacturing defect 5 type name (Max of 8 chars)
.
>>>>>>>>> Generic mfg defect rate per square foot location  (Values, Loc-range)
0.0030   0.0037   0.0005   0.0012   0.0008       001  096
.
>>>>>>>>> Manufacturing defect reduction factor per failure mode
0.89     0.92     0.57     0.68     0.97     (CS)
0.89     1.00     0.67     0.68     0.97     (TS)
.
---------+---------+--------+---------+---------+---------+---------+---------+
16       Moisture/Temperature Information                          MOITMP.INC
---------+---------+--------+---------+---------+---------+---------+---------+
Y        Usage Indicator flag (Y or N)
.
```

```
>>>>>>>>> Moisture curve count and size
  4       The number of time/moisture curves being used (Max of 10)
 35       Count of points for each curve (Max of 50)

>>>>>>>>> Moisture curve "X" coordinate data
  0    12    24    36    48    60    72    84    96   108
120   132   144   156   168   180   192   204   216   228
240   252   264   276   288   300   312   324   336   348
360   372   384   396   408

>>>>>>>>> Moisture curve thickness and "Y" coordinate data
0.0       Thickness associated with time/moisture curve 1
0.00  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90
0.1       Thickness associated with time/moisture curve 2
0.00  0.89  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90
0.2       Thickness associated with time/moisture curve 3
0.00  0.82  0.89  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90
0.3       Thickness associated with time/moisture curve 4
0.00  0.60  0.79  0.86  0.88  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90  0.90
0.90  0.90  0.90  0.90  0.90

>>>>>>>>> Temp/Mtrl Strength curve size and point information
  3       Count of points for the temp/mtrl strength curves (Max of 20)
-65.0   75.0    180.0          X1, X2, X3
1.085   0.941                        (Y1, FM 1-2, 0%)
1.000   1.000                        (Y2, FM 1-2, 0%)
0.909   0.980                        (Y3, FM 1-2, 0%)

1.085   0.941                        (Y1, FM 1-2, 100%)
0.972   1.000                        (Y2, FM 1-2, 100%)
0.838   0.980                        (Y3, FM 1-2, 100%)

>>>>>>>>> Material Strength at Room Temp, per failure mode
1.0  1.0

+--------+---------+---------+---------+---------+---------+---------+
17       Operational Defects Information                      OPRDEF.INC
+--------+---------+---------+---------+---------+---------+---------+

>>>>>>>>> Count and types of Operational Defects
  3       The number of operational defect types being used (Max of 3)
RUNWAY       Operational defect 1 type name (Max of 8 chars)
TAIL         Operational defect 2 type name
MID          Operational defect 3 type name

>>>>>>>>> Operational defect rate per location (after inspection)
NYY  3.3E-8  2.7E-8  0.8E-8      001  021
YNY  3.3E-8  2.7E-8  0.8E-8      022  043
NNY  3.3E-8  2.7E-8  0.8E-8      044  096

>>>>>>>>> Operational defect reduction factors (1 line per failure mode)
0.46  0.46  0.57        (CS)
0.67  0.67  0.67        (TS)

+--------+---------+---------+---------+---------+---------+---------+
```

```
18        OTHER Mandatory Distributions                              DSTRIB.INC
.
>>>>>>>>> Count of Skin temperature distributions blocks
1
.
>>>>>>>>> Skin Temperature Distribution Block 1
DISCRETE
14
0
-65.0    0.1231
-37.0    0.0052
-26.0    0.4111
-39.0    0.0358
-21.0    0.0347
-12.0    0.0532
-3.0     0.0254
2.0      0.0603
14.0     0.0202
29.0     0.0650
38.0     0.0493
55.0     0.0818
100.0    0.0183
160.0    0.0163
1                 Count of range specification lines
001  096          [Beg Loc, End Loc]
.
>>>>>>>>> Average moisture content percentage distribution (TRIFLEX is assumed)
0.0   1.0   0.1     Min, Max, Rho of the TRIFLEX distribution
.
00        END OF FILE
```

APPENDIX D—HISTORICAL DATA COLLECTION AND ANALYSIS

This task consisted of historical data collection and analysis. Failure data including structural failures for commercial and military aircraft was collected through visits to airlines and aviation depots. Data was probabilistically analyzed to develop historical values for aircraft probability of failure and structural probability of failure.

Contacts were made with commercial airline composite repair facilities to assess the availability of failure data. The following table summarizes the results of these contacts.

AIRLINE	LOCATION	MEETING DATE/TIME	PRINCIPAL CONTACT	COMMENTS
American	Tulsa, OK	May 17	Jim Epperson	Quantitative data plus expert opinion. Tour of repair facility.
Delta	DFW, TX	May 25	Charlie Hicks	Quantitative data plus expert opinion. Tour of repair facility.
United	San Francisco, Calif.	July 29	Murray Kuperman	Quantitative data plus expert opinion. Tour of repair facility.
De Havilland	Toronto, Canada		Kevin Ryan	Data received
NADEP	North Island, San Diego, Calif.	July 28	Guy Theriault	Quantitative data on F-18. Expert opinion. Tour of repair facility

Visits were made to commercial airline composite repair facilities to collect failure data. Additionally, data was received from De Havilland, Toronto, Canada, responding to a questionnaire.

The following summarizes information obtained from:

- American Airline Composite Repair Center, Tulsa, OK
- Delta Airline Maintenance Facility, Dallas/Fort Worth (DFW) Airport, TX
- De Havilland Inc., Toronto, Canada
- United Maintenance Facility, San Francisco, Calif.

D.1 AMERICAN AIRLINE.

Patrick Gray, Ralph Coleman, and Magdy Riskalla, Vought Aircraft Company, visited American Airline Composite Repair Center, Tulsa, OK, on May 17, 1994. Mr. J. R. Epperson, Project Engineer, American Airlines, responded to Vought's data collection questionnaire. The visit was concluded with a tour of the repair facilities. The following is a summary of information collected during the visit.

Quantitative data was obtained on:

- Fleet size
- Flight hours per year
- Induced damages

- Frequency of hail storms
- Frequency of lightning strikes

Qualitative information was obtained on

- Damage causes
- Damage location

Expert opinion was expressed on

- Airline manufacturers maintenance philosophies
- Composite structure worst enemy

Table D-1 summarizes the flight hours for each aircraft group in 1993 and table D-2 gives an associated operations event summary for those aircraft in that time period.

TABLE D-1. FLIGHT HOUR SUMMARY FOR 1993

AIRCRAFT	TOTAL FLIGHT HOURS	AVERAGE FLEET SIZE PER DAY	FLIGHT HOURS PER AIRCRAFT PER YEAR	FLIGHT HOURS PER AIRCRAFT PER DAY
B727	345,382	124.5	2774.2	7.6
B757	232,230	69.8	3327.1	9.1
B767	231,307	50.3	4598.5	12.6
MD80	769,916	251.7	3058.9	8.4
MD11	60,562	17.3	3500.7	9.6
DC10	158,253	24.5	6459.3	17.7
A300	99,848	33.3	2998.4	8.2
F100	108,398	47.4	2286.9	6.3
Total	2,005,896	618.8	3241.0	8.9

TABLE D-2. OPERATIONS EVENT SUMMARY

Average daily fleet size	618.8
No. of flight hours per year	2005896.0
Induced damages	90.0
No. hail storms (in 2,005,896 flight hours)	5.0
No. lightning strikes (in 2,005,896 flight hours)	60.0
Dropped tools	0.0
Bird strike to composite parts	0.0
A check	every 7-8 days
B check	every 30-45 days
C check	every 3-5 days

Damage causes on the ground are

- Service vehicles strike nacelles, gear doors, and baggage doors
- Baggage vehicles strike nacelles, gear doors, and baggage doors
- Catering trucks strike flaps, wing tips, and horizontal tail
- Deicing boom strikes rudder
- Fuel trucks
- Work stands both powered and unpowered
- Exhaust
- Movable surfaces with zero tolerance
- Hydraulic fluids, engine lube oil, solvents, cleaning fuel, etc.
- Water absorption

Damage causes in the air are

- Ice impacts
- Rain erosion
- Slipstream wear

General comments:

- Boeing does the most thorough job on repair manuals
- Douglas tends to put the repair manual together as design is complete
- Airbus has many vendors each with their own specs and allowables
- Defer maintenance is defined for each part in repair manuals
- Manufacturers do a poor job of sealing leading edge composite parts
- Water absorption may be significantly reduced by paint and sealers
- Manufacturers need to consider composite supportability issues early in the design stage.
- Tapping is the most widely used method of detecting delamination, ultrasonics does not work, laser shearography is used.
- Reviewing logbooks for information is an extremely time consuming task.

D.2 DELTA AIRLINE.

Delta Maintenance Facility at DFW Airport was visited on May 25, 1994. The attendees were: Vought—Ralph Coleman, Jerome Connolly, D. D. Currie, F. Stephen Beckman, James Foster, Rick Fucik, Johnny Gilliland, Sonny Gray, Magdy Riskalla, and Tom Schneider. Delta—Mike Botch and Gene Smith. This being a local trip, a large contingent of Vought engineers covering most aspects of composite issues made up the visiting team. Each one was asked to write a trip report. The following is a summary of all the information obtained during the visit.

D.2.1 Delta Overview.

There are 563 aircraft in the fleet. DFW handles 737s, 757s, 767s, and overflow on MD-80s. The 737s fly approximately 6-7 hours per day, 757s approximately 10 hours per day, and 767ERs (Extended Range) approximately 18 to 20 hours per day. Delta has three check levels: 1/2 C every 2000 to 2500 (aircraft dependent) hours, C check every 4000 to 5000 (aircraft dependent) hours and a Heavy Maintenance Visit (HMV) every 10 years. Delta performs limited repairs at their facilities in Los Angeles, Salt Lake City, Cincinnati, Tampa, Frankfurt, Atlanta, and Dallas. Major repairs and their autoclave are located in their largest facility in Atlanta. Atlanta houses their engineering facilities where more detailed information on flight hours per month, damage type, damage location, and damage frequency should be available. Down time per aircraft is typically less than 30 days per year. Block time for repairs is 14 hours. Typically, the Dallas repair facility has the aircraft for 11 hours: 1-2 hours for inspection to find any problems, 1 hour for the lead mechanic to determine the fix, and the remaining 8 hours are used to complete the repairs. The Dallas facility sees between 35 and 40 aircraft a month for 1/2 C checks.

D.2.2 DFW Facility.

Repairs on and off the vehicle. No autoclave. The largest repair possible at DFW is limited to those using a 12- x 24-in. heat blanket. Larger repairs go to Atlanta. DFW repairs fiberglass, graphite, and Kevlar reinforced composite structure. All repairs are conducted in accordance with Boeing documents.

D.2.3 Problem Areas.

The 757 and 767 leading edge slats, 2 or 3 leading edge repairs per wing, wing-to-body fairings, spoilers replaced and sent out for repair. Everything with honeycomb that is exposed to the sun (i.e., upper surfaces) is subject to getting damaged. Edge erosion problems with gear doors, etc., severity of fire damage, internal floor damage, heat sink problems, particularly with 350°F cure materials. Most repairs are to Kevlar and graphite. Control surfaces have a high repair requirement and they are prone to lightning strikes, particularly around the static wick.

D.2.4 Observations and Discussion.

Delta maintenance people feel that composites are not as durable as aluminum. They are divided on whether moisture gets into the honeycomb even when there is no damage to the part. They observed that they only have honeycomb problems on the surfaces and suspect that this may be due to solar exposure. Their means of detecting honeycomb damage (moisture ingression) is to observe the vehicle when it lands. The moisture inside has frozen and leaves condensation patches on the outer skin.

Vought feels that thin skin honeycomb is prone to foreign object damage (FOD), and it is likely undersurface honeycomb skins are sustaining some level of damage from FOD. However, since it is a lower skin, moisture will not accumulate the same as for an upper skin. We appear to need a better moisture barrier for honeycomb that will remain intact after some FOD or solar damage.

Delta personnel feel that the freeze/thaw cycles may cause as much as 50 percent of the damage they see. They did not see hail as a problem with composites even though it is a problem with aluminum. This is probably because the damage caused is in the form of delaminations, etc. and is not apparent on the surface as the dimples typical in aluminum. They feel 99 percent of all damage comes from operations with only 1 percent due to maintenance personnel errors.

The composite leading edge slat problems only occur on the outer, highly flexed slats. Delta personnel feel that composites do not handle flexing very well. Vought feels that there may be a stiffness mismatch on the outer slats causing the composites to carry a larger share of the load than they were designed to carry. Delta personnel felt that there was a lot of edge erosion problems, particularly on doors. It may be that the doors get edge damage when they are opened and closed and this then makes them more susceptible to edge erosion.

Fire damage worried Delta personnel. They observed a case where a composite part got burn damage and delaminated and plies peeled off the part. They felt that aluminum resists fire damage better than composites.

A lot of internal floor damage occurred and Delta felt that most of this was caused by ladies' high heel shoes. A thicker floor may be required or an alternative material. The food carts do not appear to damage the floors.

Except for the trailing edge on slats, Delta does not appear to have a problem with adhesive bonding. Most composites are 250°F cure which causes them big problems due to the heat sink issue, i.e., aluminum substructure conducts heat away quicker than they can add it.

D.2.5 Delta Procedures.

They use the following nondestructive inspection (NDI) techniques: x-ray, eddy current, ultrasonic, and tap test. They find the tap test to be as reliable as the more sophisticated procedures. In addition, they look for surface waviness as an indicator of damage.

Sealer is added to some of the composite panels which solve the water intrusion problem, but only for a while.

Speed tape is used to affect temporary repair. It must be inspected every 300 flight hours and must meet very strict cosmetic repair requirements. It must look exactly like surrounding structure.

For moisture removal, they drill a small hole at the lowest location on the part, apply suction through the hole and apply heat with a heat lamp. Typically, this can take 2-3 hours but can be as long as 4-5 hours for panels with high moisture contents.

They find powdery/flaky residue on composites and they think it is caused by ultra violet (UV) or other environmental agents.

Delta personnel want more lenient fly back limits for composite parts, i.e., they want a lot of it put on scheduled versus unscheduled maintenance. They are trying to collect data on composite parts so that they can establish a maintenance schedule for these parts.

Delta feels the high cost of composite repair is related to the freezer requirements, having to thaw materials out prior to using them, and tracking material total-out times. They want room temperature cure and unlimited shelf life materials.

This author feels that part of the problem is they do not do zoned repairs, they can only do very simple repairs. They do not have good thermal management systems available for repair.

They state that 15 percent of the parts on the vehicles are composites but that they require 55 percent of the total repair hours.

D.2.6 Environmental Issues.

They use methyl ethyl ketone (MEK) on a very limited basis. Mostly, they use isopropyl alcohol for a solvent. They have a paint booth and are awaiting approval of the air filtering system from the Environmental Protection Agency (EPA). Their major hazardous waste problem is from waste oil and fuel. They are trying to get approved for qualification as a small quantity (hazardous waste) generator.

D.3 DE HAVILLAND INCORPORATED.

Kevin Ryan, Customer Support Engineer, provided the following information responding to a questionnaire.

- There are 270 Dash-8 series 100 and 88 Dash-8 series 300.
- About 239 cycles per month and 1.3 hours per cycle per aircraft.
- Inspection done by visual, ultrasonic, and tap tests.
- About ten composite repairs are done at Toronto facility per month. This makes up about 10% of total Dash-8 composite repairs.
- Inherent causes about 5 percent.
- Major inherent causes are delaminations, disbands between different materials, i.e., aluminum to Kevlar, unitape to cloth, etc.
- Frequencies of maintenance and service induced damage were as summarized in tables D-3 and D-4.
- Epoxy glass is the easiest to repair.

- Unidirectional Kevlar is the hardest to repair.
- Four to six hours to repair the simplest repair.
- Twenty hours to repair the average repair.
- Forty to sixty hours to repair the hardest repair.

TABLE D-3. OPERATIONS DAMAGE, FREQUENCY, AND AFFECTED COMPONENTS

	Damage %		
	Small	Big	Component Damaged
Accidental impact with tools	15	0	Lower cowl, flight control trailing edge
Vehicles	5	10	Nose, compartments, tail
Debris blown by engine	10	0	Ice shields, fuselage
Fluids	5	0	Lower cowl
Bird strike, hail damage, replacement damage, erosion, heat damage	40	15	Leading edge, wing to fuselage fairings, flap/aileron, trailing edges, floor panels, nacelle fairings
Total	75	25	

TABLE D-4. BIRDSTRIKE, HAIL, AND LIGHTNING FREQUENCIES

	Damage per Aircraft	
	Small	Big
Bird strike	3-4 per year	1 every 1-5 years
Hail	2-3 per year	2 per year
Lightning	6 per year	2 per year

D.4 UNITED AIRLINES.

P. M. Gray and Magdy G. Riskalla visited United Airline Maintenance Facility, San Francisco, California, on 29 July 1994. The purpose of the visit was to collect data on operational damage, location, and relevant flight hours as part of the FAA Probabilistic Design Contract.

The hosting personnel were

- Murry Kuperman, Senior Staff,
- Robert De Rosa, Senior Staff,
- Eric Chesmar, Senior Engineer, and
- John Player, Senior Engineer.

A presentation was made to the hosting United Airlines personnel on Vought's probabilistic design process and on the FAA probabilistic design contract. The hosting team was extremely cooperative and provided much information on the United Airlines fleet maintenance history. Pages D-18 through D-30 contain the International Air Transport Association (IATA) questionnaire on composite structure maintenance. The IATA questionnaire represents the response of 19 worldwide small-, medium-, and large-size operators. Combined, the 19 operators maintain a total of about 2100 commuter, short-, medium-, and long-range commercial aircraft of which about 1000 aircraft have advanced composite structure. The IATA report is done in two parts with Part I summarizing main conclusions and Part II showing detail conclusions and results.

Appendix A contains Vought's questionnaire which was used as a vehicle for data collection for commercial airplanes. The questionnaire collects data on fleet information, operating environment, aircraft/composite part designation, failures of composite parts, inspection methods, and repairs. United Airlines hosting personnel responded to Vought's questionnaire which was followed by a tour of the maintenance facilities.

General information on the United Airlines fleet operation is as follows:

- Typical flight time per day is 9 hours.
- Typical ground-air-ground cycles per day is four.
- Maintenance events are 50/50 split between maintenance induced and flight damage.
- Three occurrences of bird strike in 1993.
- Hail damage about one event per year for the fleet.
- Metal-bonded structure is biggest maintenance issue.
- Kevlar is undesirable.
- Deferred maintenance is performed using 600 mph tape.

The United Airlines fleet is as follows:

A320-232	13
DC10-10	40
DC10-30	8
727-222A	75
737-222	45
737-291A	24
737-322	101
737-522	57
747-122	13
747-123	5
747SP	8
747-222B	2
747-238B	7
747-422	22
747-451	1
757-222	88
767-222	19
767-322ER	23
Total	551

D.5 NADEP, NORTH ISLAND, SAN DIEGO, CALIF.

P. M. Gray and Magdy G. Riskalla visited the NADEP at North Island, San Diego, Calif., on 28 July 1994. The purpose of the visit was to collect data on operational damage by location and relevant flight hours as part of the FAA probabilistic design contract. The North Island Depot deals mostly with the F/A-18.

The hosting personnel were

- Jerry Laibson, F-18 Air Vehicle Branch Head,
- Guy Theriault (Terio), F-18 Advanced Composite Project Leader, and
- Don Harmston, Senior Material Engineer.

A presentation was made to Mr. Laibson and Mr. Theriault on Vought's probabilistic design process and on the FAA probabilistic design contract. The hosting team was extremely cooperative and provided us with much information on the F/A-18 maintenance history. They emphasized that quantitative information on specific maintenance events with relation to composite parts would be a major task. They also expressed the view that they were under-manned and over-tasked and that they would be willing to be funded to perform this in FY 1995.

Appendix B contains the questionnaire which was used as a vehicle for data collection. The questionnaire consists of six sections; fleet information, operating environment, aircraft/composite part designation, failures of composite parts, inspection methods, and repairs. Mr. Theriault and Mr. Harmston responded to Vought's questionnaire which was followed by a tour of the maintenance facilities at NADEP, North Island, San Diego, Calif.

D.5.1 Fleet Information and Operating Environment.

The Naval Aviation Depot has estimated their total man hours needed to do the research required to answer all questions and Vought's questionnaire to be 644 hours at $82 per hour for a total of $52,808. They request that the FAA forward a funding document with the required funds.

D.5.2 Aircraft.

Figure D-1 is a major component drawing of the F-18 where shaded areas represent composite parts. Table D-5 shows composite material used in each part and the type of construction that is used.

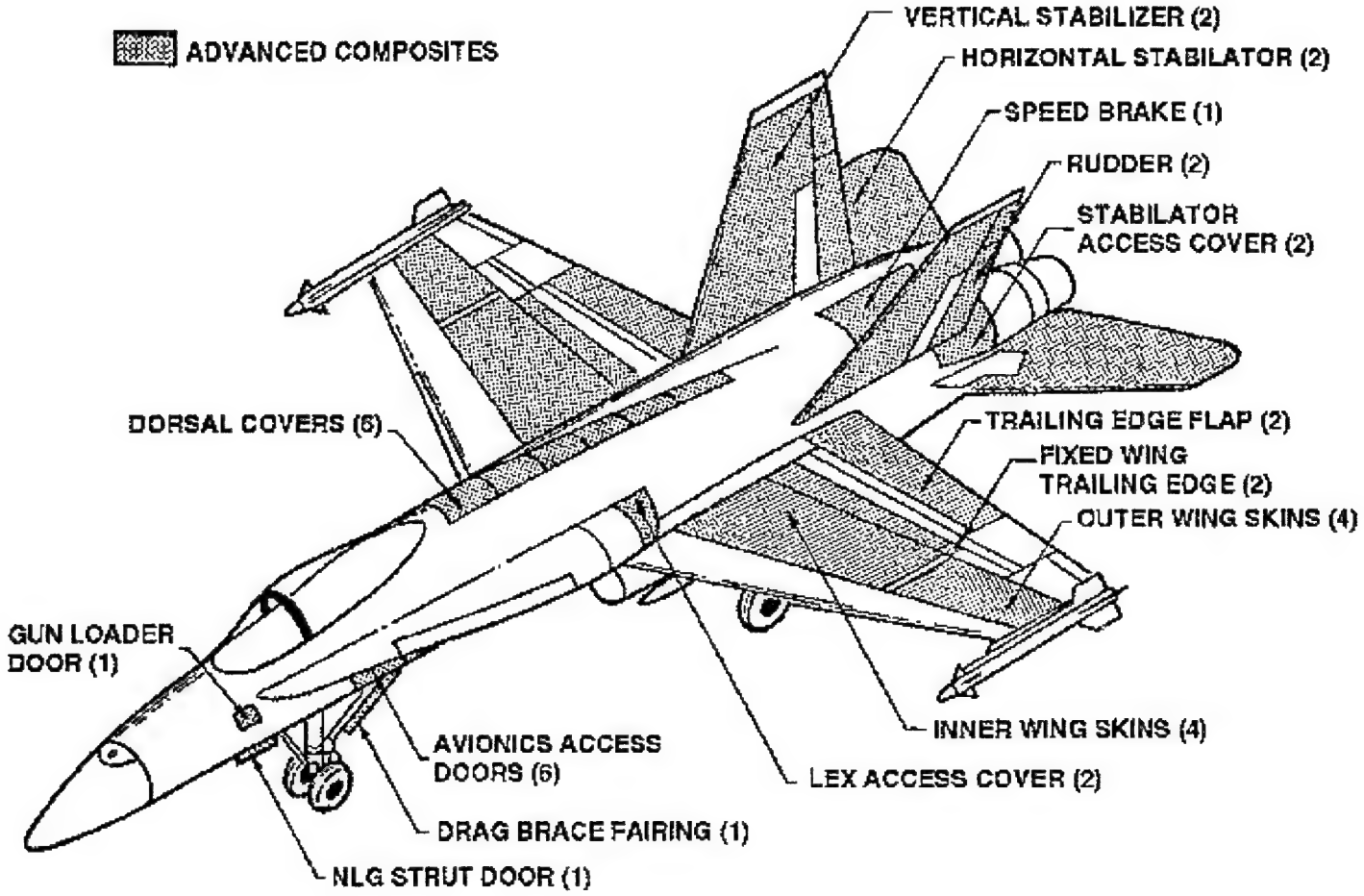


FIGURE D-1 F-18 MAJOR COMPONENT DRAWING

TABLE D-5. F-18 COMPOSITE COMPONENTS

Component	Count	Material	Construction
Vertical stabilizer	2	Carbon/epoxy	Honeycomb core monolithic structure
Horizontal stabilator	2	Carbon/epoxy	Honeycomb core
Speed brake	1	Carbon/epoxy	Honeycomb core
Rudder	2	Carbon/epoxy	Honeycomb core
Stabilator access cover	2	Carbon/epoxy	Honeycomb core
Trailing edge flap	2	Carbon/epoxy	Honeycomb core
Fixed wing trailing edge	2	Carbon/epoxy	Honeycomb core
Outer wing skins	4	Carbon/epoxy	Monolithic over aluminum spars
Inner wing skins	4	Carbon/epoxy	Monolithic over aluminum spars
LEX access cover	2	Carbon/epoxy	Honeycomb core
Avionics access doors	6	Carbon/epoxy	Honeycomb core
MLG strut door	1	Carbon/epoxy	Honeycomb core
Gun loader door	1	Carbon/epoxy	Honeycomb core
Dorsal covers	6	Carbon/epoxy	Honeycomb core

Failures of Composite Parts: Maintenance induced damage (MID) makes up the major portion of repairs. Quantitative information on each component by failure type is not really available. However, information was obtained on some aircraft components with locations of the most common damage encountered. Figures D-2 through D-5 show such locations on major composite components.

- Horizontal Stabilator:
  - MID is due to dropped tools
  - Component overlap proximity (vertical and horizontal) causes related damage
  - Scraping on fuselage is caused by parts not rigged properly
  - Tip damage is due to hits in hangar operations
  - FOD damage due to runway debris and blown tires
- Vertical Stabilizer:
  - Bird strike/FOD on leading edge and rudder
  - Dorsal deck floors at base
  - FOD damage due to runway debris and blown tires
- Trailing Edge:
  - Trailing edge flap scraping fuselage (inboard)
  - Trailing edge flap scraping aileron (outboard)

- Main Landing Gear Doors:

  - Operations/missile handling damage due to Sparrow rail immediately above doors
  - Blown tires cause damage to doors
  - Poor rigging results in internal interference damage
  - Hangar hits, rigging, and blown tires are major actors
  - Erosion damage is nil
  - Blown tire damage to wing skins (only twice for fleet so far)
  - Only 2.8% of total maintenance data is due to blown tires
  - About 0.5 tire malfunction per 1000 landings

- General Comments:

  - Not many hail or lightening strike occurrences
  - Two to three work stands/equipment under horizontal stabilator incidents per year
  - No problems with nose tires
  - Hangar hits, rigging, and blown tires are major factors
  - Erosion damage is nil
  - Blown tire damage to wing skins (only twice for fleet so far)
  - Only 2.8% of total maintenance data is due to blown tires
  - About 0.5 tire malfunction per 1000 landings

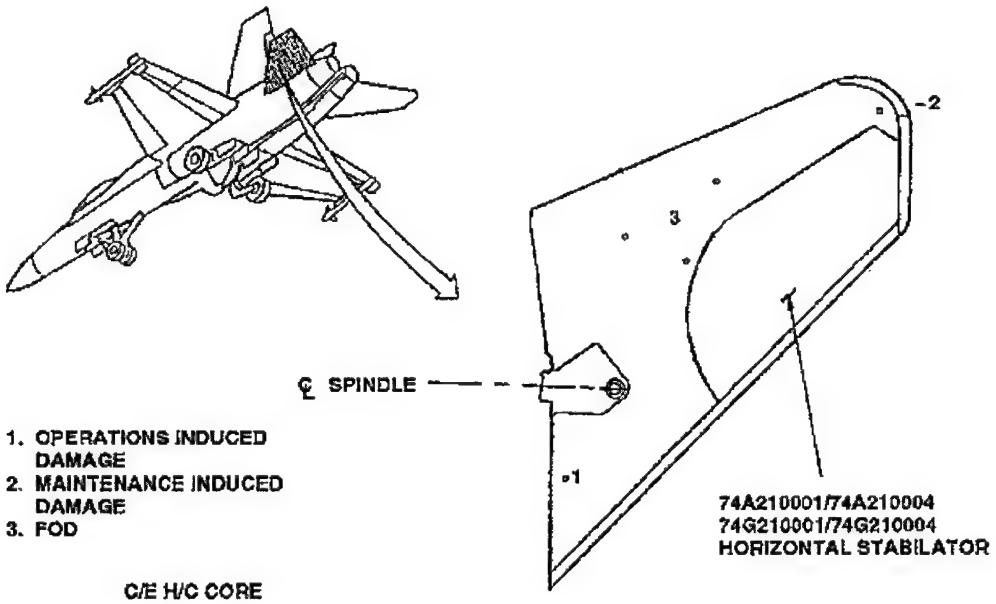


FIGURE D-2 F-16 HORIZONTAL STABILATOR DAMAGE

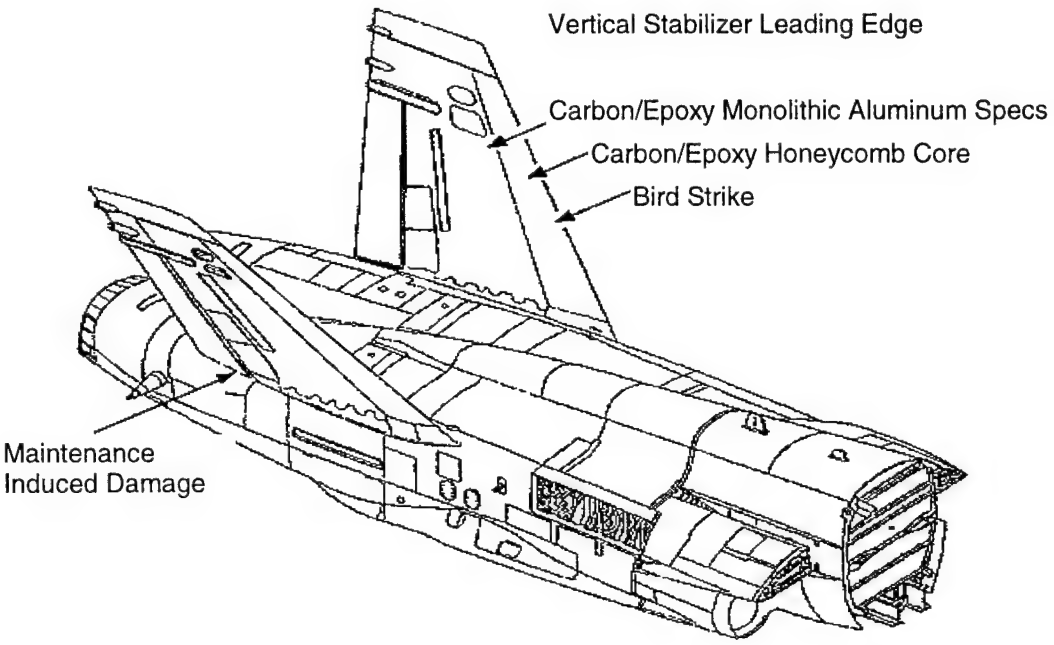


FIGURE D-3. F-18 VERTICAL STABILIZER DAMAGE

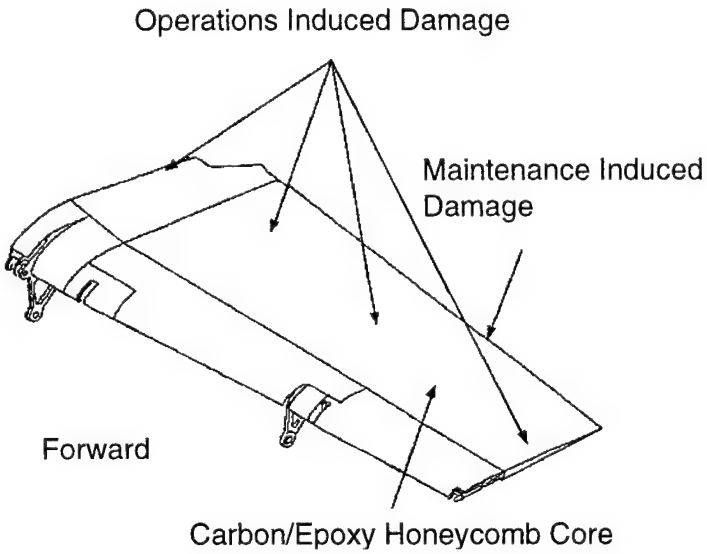
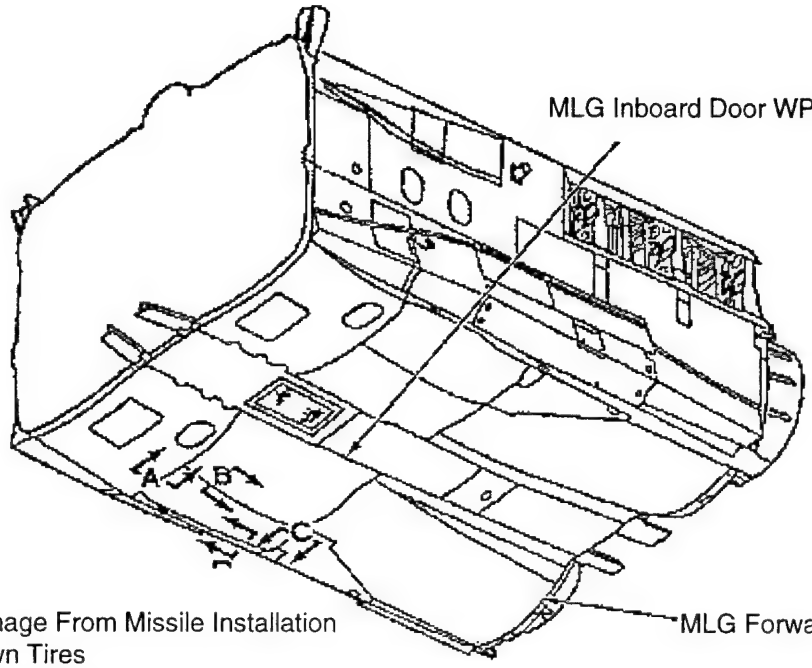
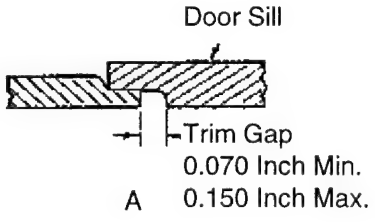
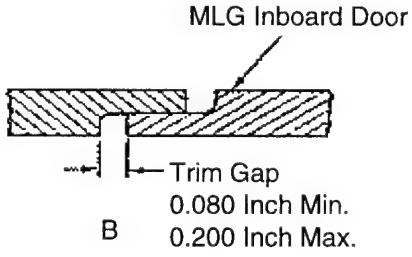


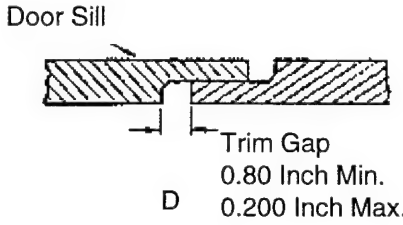
FIGURE D-4. F-18 TRAILING EDGE FLAP DAMAGE



1. Damage From Missile Installation
2. Blown Tires
3. Poor Rigging







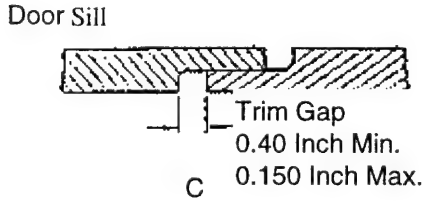


FIGURE D-5. F-18 MAIN LANDING GEAR DOOR SCHEMATIC

D.6 CONCLUSIONS OF THE IATA QUESTIONNAIRE ON COMPOSITE STRUCTURE MAINTENANCE.

PART I: OVERALL CONCLUSIONS.

These conclusions are based on the answers of 19 operators of small-, medium-, and large-size aircraft located on all continents.

The 19 operators combined, maintain a total of about 2100 commuter, short-, medium-, and long-range commercial aircraft, of which about 1000 aircraft have advanced composite structure.

The combined fleet represents all major aircraft manufacturers.

The major hurts of the operators can be regarded with respect to

- Cost
- Safety
- Maintainability
- Service history
- Future

OPERATION.

- Operator concerns regarding operational aspects of their fleet is regarded with respect to
  - Operational delay at dispatch from the gate.
  - Delayed release form maintenance check.
- Numerous operators indicate that in 70%-90% of all damages faced were the Service Repair Manual (SRM) does not give a practical solution. The main reasons are
  - Nonavailability of required repair materials.
  - Exceedance of allowable damage limits (ADL's) and repairable damage limits (RDL's).
  - Lack of time in hangar check or on the ramp to perform repair.

  Note: This also included shop repair (see also maintainability).
- Severe operational delays would follow if the SRM were strictly followed. The main reasons are
  - Repair materials (most operators state this would often or almost always be the cause of a delay).

- ADL's (most operators state this would often or almost always be the cause of a delay).

- Time to solve the problem is too short.

- Actual operational delays do not frequently occur due to a composite structure related cause. Actual delays are mostly caused by

  - No spares available.
  - Severe FOD.
  - Late damage discovery.
  - (too) long repair time.

- No actual operational problems because

  - Operators (have to) find a great deal of practical solutions.
  - Extensive temporary and/or high-speed tape fixes.
  - Mostly secondary structures are involved, no airworthiness problem.

COST.

- Cost figures of operational delays are hard to give; however, these are known to be very high.

- Composite repairs are more costly than equivalent metal repairs.

- The cost of man-hours, repair materials (also because lack of standardization), and repair equipment (in that order) is considered a problem.

SAFETY.

- Little effect on safety experienced (secondary structures).

- Majority of normal service damage exceeds allowable damage limits, however, cannot be repaired under normal operational conditions.

MAINTAINABILITY.

- Numerous operators state that of all damages occurring, 70%-90% is taped off under operational pressure until final permanent repair and exceeds allowable damage limits.

- Numerous operators indicate that in 70%-90% of all damages faced the SRM does not give a practical solution. The main reasons are

- Nonavailability of required repair materials.

- Exceedance of allowable damage limits (ADL's) and repairable damage limits (RDL's).

- Lack of time in hangar check or on the ramp to perform repair.

Note: This also included shop repair.

SERVICE HISTORY.

- Glass fiber-reinforced composite structure repairs account for 90% of total volume. Advanced composite structure repairs account for 10% of total volume.

- The 10% advanced composite structure repairs cause most of the problems.

FUTURE.

- Increase use of composite structures.
- More complex primary applications.
- Impact on operation, cost, safety, and maintainability will also increase.
- Maintenance of composite aircraft structure must improve.

D.7 PART II: DETAILED CONCLUSIONS AND RESULTS.

1. AIRLINE FLEET, FACILITY, AND PROCEDURAL INFORMATION.

QUESTION 1

How many aircraft do you maintain?

Answer:

The combined 19 operators maintain a total of about 2100 commuter, short-, medium- and long-range commercial aircraft, representing all major aircraft manufacturers.

QUESTION 2

How many aircraft with carbon fiber-reinforced plastic (CFRP) structures do you maintain?

Answer:

The combined 19 operators maintain a total of about 1000 commuter, short-, medium-, and long-range commercial aircraft with advanced composite structure, representing all major aircraft manufacturers.

QUESTION 3

Indicate whether you have an autoclave, oven, or hot bonding units (include size).

Answer:

The following repair facilities were found:

- Four operators have an autoclave facility.
- Two operators are planning an autoclave facility.
- Ten operators have ovens from 2 up to 7 m length.
- Fifteen operators use hot bonding units (heat blanket/vacuum bag).

QUESTION 4

Do your personnel who repair exterior composite structures also repair interior panels?

Answer:

Generally, composite repair shop workers repair both exterior and interior composite structures. Some operators indicated a trend towards structure shops with multidisciplined workers for structural repairs on both metal and composite structures.

QUESTION 5

Do your aircraft mechanics use information directly from the Structural Repair Manual (SRM) or do they use airline specific procedures?

Answer:

Nine operators indicated that the aircraft mechanics use both SRM and operator specific procedures. Six operators indicated that aircraft mechanics use information directly from the SRM.

QUESTION 6

Do your aircraft mechanics consequently use the SRM to make go versus no-go and repair versus replace decisions?

Answer:

Most operators indicated that the aircraft mechanic uses the SRM for decision making. If damage exceeds allowed limits, most operators indicated that final decisions are made by engineering.

Remarks:

- One operator stated that if the damage is noticed on the ramp, engineering decides.
- One operator stated that only in 10%-30% of all go versus no-go decisions is the SRM consulted.
- One operator stated that for all cases engineering decides for go versus no-go.

QUESTION 7

Does the composite shop worker use information directly from the SRM or from the airline specific procedures?

Answer:

Eight operators indicated that the composite shop worker uses both SRM and operator-specific repair procedures. Four operators indicated that the composite shop worker uses SRM repair procedures. Four operators indicated that the composite shop worker uses only operator-specific repair procedures.

QUESTION 8

If the SRM does not cover your practical damage problem, how do you arrange for approved data to repair the damage, e.g., design of a repair authorized by local authority, approval of proposal by the Designated Engineering Representative (DER) at the Original Equipment Manufacturer (OEM).

Answer:

Most operators indicated that the approved data is arranged for via the DER at the OEM.

Remarks:

- Operators indicated that OEM's normally react quickly to operator's requests but state that frequency of requests are high.
- Some operators use local civil aviation authorities to approve repairs.
- Two operators use in-house authorities to approve repairs.

QUESTION 9

Do you keep records of composite repairs? If yes, which information do you store? Please specify which components, quantify damage size, type location, and repair method (i.e., resins, cores, fibers, cure time, etc.).

Answer:

Most operators keep records of composite repairs. Records are kept in cases of

- structural components,
- repairs that exceeded SRM limits,
- major structural repairs, and
- repairs on specific components such as flight controls and nacelles.

2. AIRLINE REPAIR PRACTICES.

QUESTION 10

Estimate how many temporary and permanent (on-aircraft and in-shop) repairs you perform on a yearly basis on glass fiber-reinforced plastics (GFRP) and advanced composite structures, including:

- temporary repairs
- permanent repairs on aircraft
- permanent repairs in shop

Answer:

Although the number of repairs per year differ largely from operator to operator, the following conclusions can be drawn:

- About 90 percent of all composite repairs are done on GFRP structures.
- Only 10 percent of all composite repairs are done on advanced composite structures.
- On GFRP structures, a substantial amount of on-aircraft repairs take place.

In general, operators have many years of experience with glass fiber reinforced composite structure repairs but only limited experience with advanced composite repairs. The advanced composite repairs represent about 10 percent of the composite repair volume, although about 90 percent of all problems.

QUESTION 11

Which temporary repairs are used and how are these joined to the damaged structure? Some examples are

- Metal doubler (Al, Ti, CRES) or precured composite doubler, either cold bonded, hot bonded, riveted, bolted, or a combination
- Room temperature (RT) wet lay-up (WLU)
- Other

Answer:

Most common temporary repairs use Al, Ti, or CRES doublers, either riveted or bolted, and sometimes bonded and sealed as well. Other regularly used repairs are RT WLU repairs (11 operators) and precured patch repairs (four operators). Several operators stated they prefer not to perform temporary repairs, because this means double work and in most cases enlargement of the damaged area by fastener holes or tear down of temporary repairs before permanent repair.

QUESTION 12

Where do you perform your temporary repairs (field, homebase, ramp, hangar)?

Answer:

There are large differences between operators with regard to where temporary repairs are performed. Temporary repairs are performed in the field, at an outstation, on the ramp, or in the hangar.

QUESTION 13

Estimate how frequently, as a percentage of all permanent repairs done, you use the following permanent repair methods:

- Room temperature (RT) wet lay-up (WLU)
- Elevated temperature (ET) WLU
- Prepreg (PP) 120°C (250°F) cure
- Prepreg (PP) 175°C (350°F) cure
- Other

Answer:

The majority (80 percent) of all permanent repairs (of both glass fiber-reinforced and advanced composite structure) are performed with WLU techniques. The use of prepregs is limited; in most cases prepreg 120°C (250°F) cures are used. Prepreg 175°C (350°F) cures are rarely used for repair purposes.

Regarding the use of ET WLU repair, it became apparent that some operators mean RT cure resins cured at elevated temperature at 60-90°C (140-195°F) to speed up the cure. Other operators mean ET cure resins that do not cure at RT and require cure temperatures in the range of 95-120°C (200-250°F).

QUESTION 14

Which permanent repairs do you perform on aircraft?

Answer:

For on-aircraft permanent repairs WLU repair is most widely used (16 operators). Some operators prefer prepregs for on-aircraft repairs.

Remarks:

- One operator indicated that prepregs are easier to handle than WLU systems in a hangar environment.
- Another operator names the disadvantage of the loss of crucial time in taking the prepregs out of the storage facility and allow the prepregs to warm to RT.
- Two operators indicated that heat and vacuum may not always be easy to apply in the hangar.

In general, on-aircraft repairs should be performed quickly enough to prevent necessary disassembly. On-aircraft repairs are generally less extensive than shop repairs. Normally, in-service damage is repairable with on-aircraft repairs during available downtime for scheduled maintenance checks.

QUESTION 15

Permanent or temporary repairs cannot always be performed. Estimate how often, as a percentage of all damages occurring, damage greater than allowable is taped off with aluminum foil tape under operational pressure.

Answer:

Large differences existed among the operators polled, who indicated either low (0.05% to 5%) or high (up to 90%) percentages. The operators giving high percentages accordingly regard the size of allowable damage as a very important (score 4) or the most important (score 5) because of operational delays (see Question 24). Operators further indicated that in practice, the deferment of repairs on damage exceeding allowable limits depends on the structural importance of the damaged area, damage size, and the availability of a spare part.

Remarks:

The low percentages given by some operators may possibly be explained by the fact that not all organization disciplines, especially hangar maintenance, were involved or data was not readily available. This statement was confirmed by several operators.

QUESTION 16

Estimate the average time interval taken before a damage, which has been taped off with aluminum foil tape or temporarily repaired, can be permanently repaired.

Answer:

There were large differences in the answers given. It is not realistic to define an average. Deferment of permanent repair ranges between one flight for severe damage up to several months for small cracks on secondary structure. Deferment of permanent repair was given a low priority as a cause of operational delays (see Question 24). This means that in most cases the operator is able to make some kind of temporary repair. Time to permanent repair mainly depends on

- structural importance of the damaged part,
- damage size (safety aspect),
- availability of repair,
- availability of spare part,
- time to next scheduled maintenance check, and
- operator policy.

QUESTION 17

Which permanent repair method do you prefer in the shop?

Answer:

- In most cases, 250°F (120°C) cure prepregs (without autoclave cure!) are preferred.

- Some operators preferred the use of prepregs with the use of an autoclave facility.

- Some operators preferred WLU, mostly because of the concerns listed below which are associated with the use of prepregs.

- Vacuum bag cure is preferred for repair purposes.

Advantages of the use of prepregs that were mentioned include ease of handling, defined material quality, better rapidity of repair in some cases, less dependence on skill of worker, better repair performance, and larger repair tolerance limits. Concerns that were mentioned: material cost,

availability of small batches, incoming material control, shelf life limitation, drapeability, and lack of standardization.

QUESTION 18

In which case, by which method, and how long do you dry composite structures prior to repair?

Answer:

All operators use drying before repair, although each operator uses his own drying procedure. Most composite structures are dried for several hours, ranging from 1 hour up to 24 hours, at temperatures between 50-70°C (120-160°F). Drying is mostly performed in an oven. Sometimes infrared lamps are used.

3. SERVICE EXPERIENCE.

QUESTION 19

Estimate, in percentages, how often each of the following damage types occur on your composite structures:

- holes
- delamination
- crack
- other

Answer:

The following damage types occurred on average:

- Holes 35%
- Delamination 45%
- Cracks 10%
- Other 10%

These data have large spreads; however, this list is at least an indication of the damage types that have to be repaired.

Remarks:

- There is a terminology problem with disbond and delamination.

- Other damages relate mostly to erosion and heat deterioration.

- Because delaminations are frequently found around hole edges, some operators also classified holes as delamination.

- There is no distinction drawn between glass fiber-reinforced and advanced (carbon- and boron-reinforced) composites. However, causes and types of damage depend mostly on the type of component and the location on the aircraft.

QUESTION 20

Estimate for each damage type named, how often, in percentages, the given damage sizes occur?

- holes: < 1.5 inches, 1.5-3.0 inches, > 3.0 inches
- delaminations: < 1.5 inches, 1.5-3.0 inches, > 3.0 inches
- cracks: < 1.5 inches, 1.5-3.0 inches, > 3.0 inches

Answer:

The numbers in the following table are calculated from the operator responses:

Damage Type and Size		Minimum Number (%)	Average Number (%)	Maximum Number (%)
Holes	< 1.5 inches	15	50	100
	1.5-3.0 inches	0	35	80
	> 3.0 inches	0	15	50
Delaminations	< 1.5 inches	0	10	40
	1.5-3.0 inches	0	30	80
	> 3.0 inches	10	60	100
Cracks	< 1.5 inches	0	30	90
	1.5-3.0 inches	0	30	80
	> 3.0 inches	0	40	100

Remarks:

- Some operators indicated that delaminations smaller than 3 inches are difficult to detect under normal inspection conditions.

- This data concerns actual in-service damage sizes. Damage up to 3 inches can be regarded as normal in-service damage. It was experienced that this in-service damage, in most cases, is beyond the applicable allowable damage limit.

QUESTION 21

Estimate which of the following damage causes has led to your composite structure damages.

- lightning; bird strike, hail; moisture, chemical; runway stones; platform servicing, hangar maintenance; other

Answer:

The following damage causes occur on average:

• Lightning	7%
• Birds, hail	8%
• Moisture, chemical attack	30%
• Runway stones	8%
• Platform damage, hangar maintenance	36%
• Other	11%

Remarks:

- The volume of human induced damages is striking. Several operators indicated percentages as high as 50% to 70%.

- The severity of the various types of damage is not accounted for; for example, lightning strike can cause severe damage but is very infrequent as a source of damage. However, it appears that some operators are unfortunate enough to have major problems with lightning strike.

- There exists a large spread in answers for moisture as a damage cause. It ranks from rather low (10%-20%) to rather high (60%-80%). It is thought that the high percentage answers have accounted for moisture as a secondary damage which will cause additional induced damage after primary foreign object damage (FOD).

- "Other" damage causes included design deficiencies, type disintegration, overheating, incorrect installation, or "unknown."

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QUESTION 22

Have you experienced delays and/or late releases because of composite structure damage? Please specify the reasons and give typical examples if possible: e.g., spare was not readily available, structure was damaged late in check, damage was discovered late in check, repair time too long.

Answer:

Delays are not encountered frequently. This is mainly because the operators make extensive use of temporary repairs. When delays occur they are mainly caused by severe FOD. The causes of delays listed included:

- No spare available
- Late discovery of damage
- Damage caused late in the check
- Long repair time

4. AIRLINE CONCERNS.

QUESTION 23

The Structural Repair Manual (SRM) offers repair solutions for damages. The SRM solution does not apply if, for example, there is not enough time to do the repair or the damage (size, type, location) is not covered by the SRM. Estimate how often the repair solution offered by the SRM does not apply to the damages in your operational situation as a percentage of all damage faced.

Answer:

Here also exist large differences of opinion between operators. Six operators gave a low percentage (5%-10%), five operators gave a high percentage (70%-90%), and five operators gave an intermediate percentage in the range of 30%-60%. Those operators that gave low percentages are generally small operators, while operators that gave high percentages are generally larger operators. Several operators presented precise figures on the reasons for this mismatch between the situations considered in the repair manual and actual repair problems.

Remarks:

- The operators who gave a high-percentage answer to this question also indicated a high percentage of damage that needs to be taped off due to operational pressure (see Question 15).

- Operators that gave high-percentage answers also tend to give high-percentage answers to Question 24 on potential operational delays.

- The main reasons mentioned for failure of the SRM to apply include lack of time, exceedance of allowable and of repairable damage limits, and the nonavailability of repair materials.

- Three operators clearly indicated that the OEM repair solutions are more applicable for glass-reinforced composites than for advanced composite structures.

QUESTION 24

Indicate how often repair delays cause operational delays of any kind (from ramp, hangar), if the SRM is strictly followed.

Answer:

The nonavailability of specified repair materials, low allowable damage limits, and excessive total repair time are the major causes.

QUESTION 25

Give a priority ranking from "most important" to "least important" of operator concerns with regard to maintenance of composite structure.

Answer:

No information available.

QUESTION 26

Which effects do problems with composite structure inspection and repair have on (1) airworthiness, (2) operations, and (3) cost in your organization? Please explain per aspect and/or give examples.

Answer:

The operators indicated the following effects on airworthiness/safety, operations, and cost.

Airworthiness/safety:

- Little effect. Up to now there is no extensive use of composites in primary structure.

Operations:

- Few actual delays experienced.
- If repair manuals were strictly followed, severe delays would follow.

Cost:

- Composite repairs are more costly than metal repairs.
- The cost of man-hours, material, and equipment (in that order) is considered a problem.

QUESTION 27

What is your main concern? The current problem with composite structure maintenance or the anticipated future problems with increased use of composites in your fleet.

Answer:

No available information.

QUESTION 28

Indicate how important you consider the work of each of the six proposed Commercial Aviation Composite Repair Committee (CACRC) Task Group efforts.

Answer:

No information available.

5. IMPROVEMENTS.

QUESTION 29

Is your airline willing to keep records of which damages occur and how these are handled, in order to be able to confront the OEM with specific practical data, provided that this would lead to improvements in the repairs offered by the OEM?

Answer:

There is general willingness to keep records on specific components. But this needs to be advantageous for operators in the end, e.g., by extending allowable and repairable damage limits. The record keeping system should be realized at minimal operator cost, with standardized terminology and form layout. A registration form should be easy and quick to fill in.

QUESTION 30

General statements that repairs need to be quick and that large damages can be repaired are not specific enough for the OEM. The airlines should specify the conditions under which inspections and repair can be realized in the regular check intervals. In this way, future repairs can be designed to fit better with the airline specified conditions. Do you agree with this viewpoint, and if so, are you prepared to specify your inspection and repair conditions?

Answer:

The majority of operators agreed to the need of inspection and repair conditions defined by the operators.

QUESTION 31

If all composite structure components were classified in accessible classes and zones (grading), on a picture in the manuals, the structural importance of an area on a component could be easily determined. This in turn could speed up damage assessment and subsequent action to be taken. Please comment on this.

Answer:

The operators unanimously agreed on this subject. Clarity of explanations given in repair manuals is crucial when repairing composite structures. The relationship between importance of a given area of a structure and the required inspection/repair actions must be transparently presented in the manual. A class/zone or grading system would support this.

Working with manuals that do not provide this transparency is a serious problem to all operators.

QUESTION 32

The SRM does not distinguish between different levels of repair shops. It also does not distinguish between different levels of composite repair personnel. If composite repair shops and personnel would be classified, high-level repair shops could possibly be authorized to perform more extensive repairs than lower level shops. Please comment on this.

Answer:

There is general agreement on the need for classification of repair shops if it leads to the capability for more extensive repairs. Some small operators were not enthusiastic about classifying repair shops. However, classification seems desirable, as skill and equipment factors strongly determine the repair performance.

Remarks:

- Two operators suggested that the CACRC should define this classification.
- Classification would enable operators to value investments in training, facilities, and equipment.

QUESTION 33

Sometimes locally engineered repair solutions work well for specific problems. Are you prepared to submit these solutions to the CACRC in order to possibly establish wider usage of the method?

Answer:

Eight operators are prepared to submit locally engineered repairs to the CACRC in order to share and extend the knowledge and experience involved with composite structure repair. One operator mentioned liability as being a potential problem.

QUESTION 34

Give your priority on where the effort towards better maintainable structures should start.

Answer:

Most operators indicated that standardization of repair materials, practices, procedures, and processes should be the highest priority. Also, several operators mentioned design of composite structures for cost effective maintenance or design for maintainability as their priority.